NASA CR-

Computer Sciences Corporation

CSC/SD-77/6017 UD1

CSC

(NASA-CR-183448) APPLICATIONS EXPLORER MISSIONS-A/HEAT CAPACITY MAPPING MISSION (AEM-A/HCMM) ATTITUDE SYSTEM FUNCTIONAL SPECIFICATIONS AND REQUIREMENTS (Computer Sciences Corp.) 386 p

N90-70152

Unclas

00/43 0234835

CSC/SD-77/6017UD1

APPLICATIONS EXPLORER MISSIONS-A/HEAT CAPACITY MAPPING MISSION (AEM-A/HCMM) ATTITUDE SYSTEM FUNCTIONAL SPECIFICATIONS AND REQUIREMENTS

(Revision 1)

Prepared for

GODDARD SPACE FLIGHT CENTER

By

COMPUTER SCIENCES CORPORATION

Under

Contract NAS 5-11999
Task Assignment 542

Prepared by:

R. Byrne 5/15/78
Dr. R. Byrne Date
Task Leader

J. S. Legg, Jr.
J. W. Wood
Dr. S. G. Hotovy
Dr. D. Niebur
Dr. P. J. Camillo
R. S. Nankervis, GSFC

Approved by:

R M Lerner 5/15/78
Dr. G. M. Lerner Date
Section Manager

C. B. Spence 5/15/78
Dr. C. B. Spence Date
Quality Assurance Reviewer

M Joseph 5/15/78
M. Joseph Date
Department Manager

ABSTRACT

This document describes the Attitude Support System required for attitude determination and control of the Applications Explorer Missions-A/Heat Capacity Mapping Mission (AEM-A/HCMM). It includes descriptions of the mission profile, the relevant spacecraft hardware, the operational environment for the proposed software and the capabilities and features to be provided by the attitude support system. Section 2 was written by the Attitude Determination and Control Section (ADCS) at Goddard Space Flight Center (GSFC). This document was updated in February 1978 to incorporate all changes to the specifications which were issued prior to January 26, 1978. The last change, Modification Number 27, requires that the Sentinel Record be recognized by the attitude determination software. This updated document will serve as the basis for the AEM-B/Stratospheric Aerosol and Gas Experiment (SAGE) attitude software specifications. Revised text is indicated by a vertical line in the margin.

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

Section 3 (Cont'd)

TABLE OF CONTENTS (Cont'd)

Section 5 - Functional Specifications

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Cont'd)

LIST OF TABLES

SECTION 1 - INTRODUCTION

The Heat Capacity Mapping Mission (HCMM), depicted in Figure 1-1, is the first in a series of satellites utilizing a basic, modularly designed launch vehicle and satellite support system called the Applications Explorer Missions (AEM). The purpose of the AEM is to provide a means of performing various tasks which require low cost, quick reaction, and relatively small spacecraft in dedicated orbits. In supporting the HCMM project, hardware designed for previous missions has been modified when possible for current use; in particular, programs such as International Ultraviolet Explorer (IUE), International Monitoring Platform (IMP), Radio Astronomy Explorer (RAE), and NIMBUS provided equipment suitable for the HCMM.

Section 1 presents a profile of the HCMM mission, as well as a description of the hardware and control system to be flown in support of the mission. Section 2 details the mission requirements as set forth by the Attitude Computations Analyst (ACA) at Goddard Space Flight Center (GSFC). Section 3 presents the capabilities specified for the attitude determination software system in support of the requirements included in Section 2. Section 4 contains analytical considerations and algorithms suggested for providing the capabilities presented in Section 3. Section 5 presents specifications for the attitude determination and control system to support the HCMM. Section 6 includes existing software suitable for selected specifications in Section 5. Section 7 describes a telemetry and attitude data simulator for system testing and analytical investigations, and Section 8 specifies utility programs which are included in the support system.

1.1 MISSION PROFILE

The purpose of the HCMM is to conduct thermal mapping of the North American continent, particularly the United States, with primary emphasis on applications related to Earth resources' availability. The heat capacity of a portion of

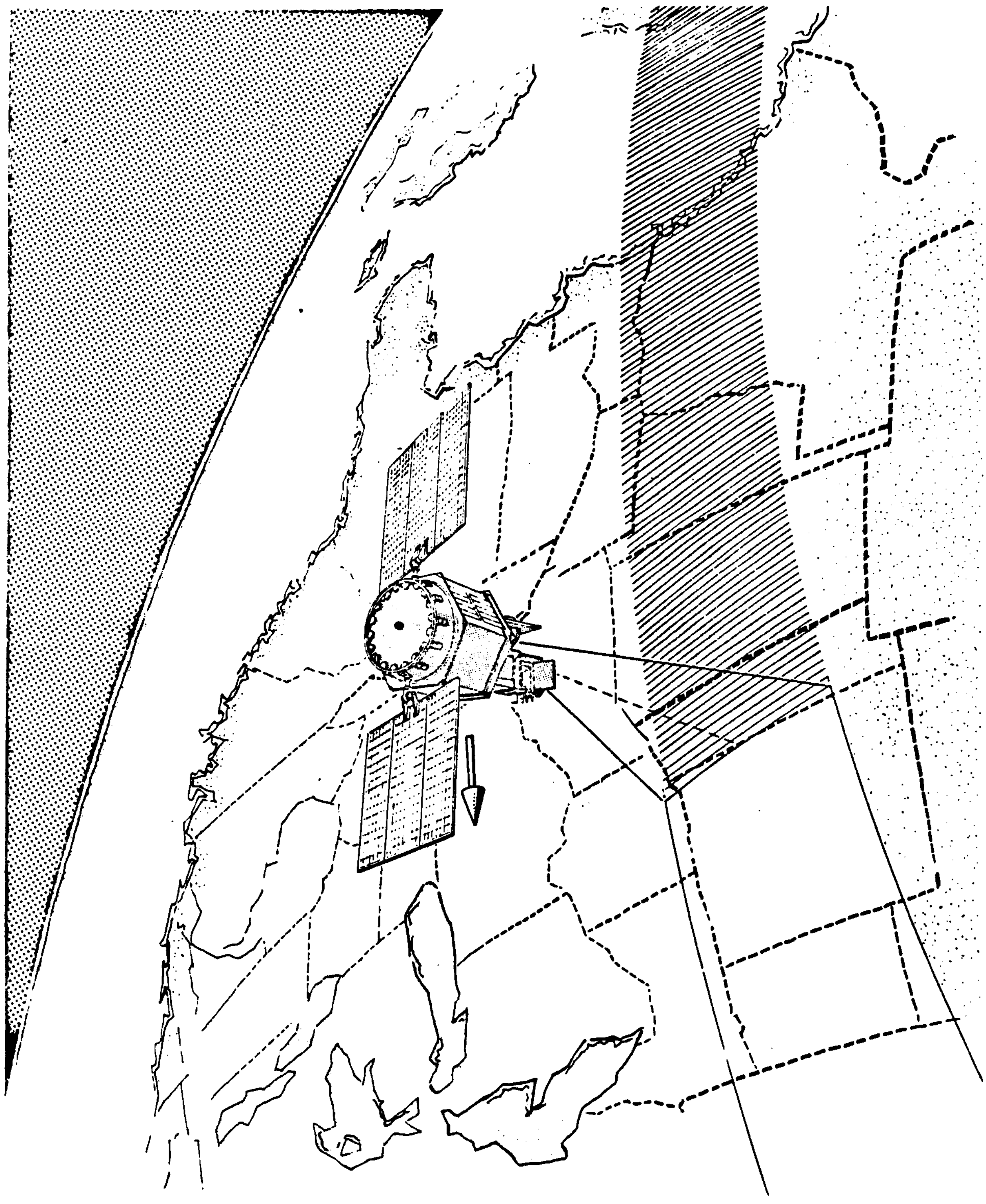


Figure 1-1. HCMM Spacecraft

the Earth's surface, together with its density and thermal conductivity, determines the rate at which temperature variations occur in the area. The extent to which the local environment tends to maintain a constant temperature is referred to as thermal inertia. This property has a profound effect on vegetation, and knowledge of the thermal inertia of a region can be used in the determination of many environmental factors influencing the area, such as diurnal temperature variation and the local weather. The optimum times for sensing temperatures in this context are approximately 1:30 p.m. and 2:30 a.m., when local temperature tends to reach minimum and maximum values.

Measurements will yield results such as the effects of cold fronts passing over an area, the extent of evaporative cooling after rainstorms, the amount of transpiration of water from vegetation, and the identification of subsurface rock types as a guide to resource availability.

1.1.1 Scientific Objectives

The scientific objectives of the mission include research into the feasibility of the following:

- Producing thermal maps at optimum times for thermal inertia measurements for discrimination of rock types and mineral resource location
- Measuring vegetation canopy temperatures at frequent intervals to determine the transpiration of water and plant stress
- Measuring soil moisture effects by observing the temperature cycle of soils
- Mapping thermal effluents, both natural and manmade
- Utilizing frequent coverage of snow fields for water runoff prediction

The orbit of HCMM is chosen to sense surface temperatures near the maximum and minimum of the diurnal cycle. All surfaces exposed to variation of atmospheric temperature and of solar illumination vary in temperature, the amplitude of variation depending on the atmospheric and solar effects and the thermal inertia of the surface. By sensing the temperature at maxima and minima during normal weather conditions, the relative thermal inertia of the surface in various areas can be determined.

The 600-kilometer (324-nautical-mile) circular, Sun-synchronous[1] orbit will have an inclination of approximately 98 degrees and an ascending daylight node with a nominal equatorial crossing time of 2:00 p.m. (local Sun time) and will thus provide 1:30 p.m. and 2:30 a.m. crossing times over middle U.S. northern latitudes. This orbit is near optimum for sensing surface thermal effects and allows reflectance measurements during daylight.

1.1.2 Scientific Payload

The device used to support the experimental objectives of the mission is a Heat Capacity Mapping Radiometer (HCMR), obtained by modifying the spare flight radiometer from NIMBUS-5. The HCMR has a very small geometric field of view (less than 1-by-1 milliradian), high radiometric accuracy, and a ground swath coverage wide enough to provide examination of selected areas within 12-hour periods containing the upper and lower diurnal limits of temperature. The instrument operates within two frequency bands, 10.5 to 12.5 microns (infrared) and 0.8 to 1.1 microns (visible).

The following four major subassemblies compose the HCMR:

- Scan mirror and drive assembly
- Optics assembly
- Electronics assembly
- Radiant cooler assembly

---

[1] "Sun-synchronous" means that the plane of the orbit precesses about the Earth at the same rate that the Earth moves about the Sun in its orbit.

The scan mirror drive assembly provides cross-track scanning of the field of view (FOV) of the HCMR as the footprint of the spacecraft moves along the ground. The optical assembly provides ground resolution and spectral selection of the two available frequency bands. The electronics assembly contains data amplifiers, housekeeping management components, and analog-to-digital conversion (ADC) equipment. The radiant cooler assembly maintains the instrument operating temperature at approximately 115 degrees Kelvin (K). The angular momentum of the scan mirror is offset by a separate motor driving a compensation mass in a direction opposite to that of the mirror.

1.1.3 Mission Time Line

HCMM is scheduled for launch from Vandenberg Air Force Base in the spring of 1978. The nominal orbital parameters are as follows:

- Semimajor axis = 6978 kilometers
- Eccentricity ≤ 0.001
- Inclination = 97.79 degrees

The Scout-F launch vehicle will consist of the four stages listed in Table 1-1.

Table 1-1. HCMM Launch Vehicle Description

	FIRST STAGE	SECOND STAGE	THIRD STAGE	FOURTH STAGE
NAME	ALGOL III	CASTOR II	ANTARES II	ALTAIR III
GUIDANCE METHOD	JET VANES AND AERODYNAMIC TIP SURFACES	HYDRAZINE REACTION JETS	HYDRAZINE REACTION JETS	SPIN STABILIZED

The sequence of events following launch is described in Table 1-2. An Orbit Adjust System (OAS) is used to trim and circularize the orbit. Its capabilities are summarized in Table 1-3.

1.1.4 Modes of Spacecraft Operations

There are three primary modes of operation of the spacecraft. The first, attitude acquisition, occurs during approximately the first six orbits. This mode may subsequently be required in the event that a hardware or software malfunction causes a large attitude error (greater than 10 degrees). The second mode, the orbit adjust phase, is scheduled to last for 1 to 2 weeks. Periodic orbit adjust maneuvers may occur throughout the life of the mission, but are not currently planned. Finally, the mission mode is the operational mode of the spacecraft, and is nominally maintained for the remainder of the mission. These modes of operation are described below.

1.1.4.1 Attitude Acquisition Mode

After separation of the spacecraft from the fourth stage of the launch vehicle, a yo-yo despin device[1] is deployed and the spacecraft despins from approximately 150 revolutions per minute (rpm) to less than 2 rpm about the Z-axis (yaw axis). Solar paddles are deployed and a momentum wheel, which contains an infrared Earth horizon sensor, is spun up to an initial 800 rpm, providing angular momentum along the pitch axis to stabilize the roll and yaw angles (Figures 1-2 and 1-3 define the pitch, roll, and yaw axes in the orbital and the spacecraft coordinate system, respectively). An autonomous onboard control loop drives magnetic coils to transfer momentum from the Z-axis to the Y-axis (pitch axis), and the momentum wheel provides a reference attitude about the pitch axis and inhibits motion about roll and yaw. The control loop drives the

[1] A yo-yo despin device is a pair of weights attached to cables which upwrap and release to decrease the angular momentum of the spacecraft.

Table 1-2. Sequence of Events Following Launch of HCMM (1 of 2)

TIME	EVENT
T–120.0 (SECONDS)	ARM PYROTECHNIC DEVICES
T	FIRST STAGE IGNITION/LIFTOFF
T + 81.7	FIRST STAGE BURNOUT
T + 84.2	SECOND STAGE IGNITION/FIRST STAGE SEPARATION
T + 123.6	SECOND STAGE BURNOUT
T + 139.1	THIRD STAGE IGNITION/SECOND STAGE SEPARATION
T + 168.0	THIRD STAGE BURNOUT
T + 562.8	INITIALIZE SPINUP
T + 587.1	FOURTH STAGE IGNITION/THIRD STAGE SEPARATION
T + 620.4	FOURTH STAGE BURNOUT
T + 913.1	FIRE PYROTECHNIC DEVICES/SPACECRAFT SEPARATION FROM FOURTH STAGE
T + 923.1	YO-YO RELEASE/DESPIN SPACECRAFT
T + 933.1	DEPLOYMENT OF SOLAR ARRAYS
T + 965.1	ON-ORBIT MODE ACQUISITION (SPACECRAFT INITIALLY TUMBLING, NO SUN DATA)
T + 975.1	WHEEL SPINUP (NO SUN DATA)
T + 34.7 (MINUTES)	ACQUISITION OF SUNLIGHT
T + 74.7	ACQUISITION OF SIGNAL (WINKFIELD)
T + 167.7	ACQUISITION OF SIGNAL (WINKFIELD; SUN DATA AVAILABLE, TUMBLING)
T + 194	ACQUISITION OF SIGNAL (ALASKA; NO SUN DATA, TUMBLING)
T + 265	ACQUISITION OF SIGNAL (WINKFIELD; SUN DATA AVAILABLE, TUMBLING, THIRD ORBIT)
T + 280	ACQUISITION OF SIGNAL (ALASKA; SUN DATA AVAILABLE, TUMBLING)

Table 1-2. Sequence of Events Following Launch of HCMM (2 of 2)

TIME	EVENT
T + 375	ACQUISITION OF SIGNAL (ALASKA; SUN DATA AVAILABLE, TUMBLING, FOURTH ORBIT, COMMAND ACS ON)
T + 407	ACQUISITION OF SIGNAL (ORRORAL; NO SUN DATA, STABLE CONFIGURATION)
TBD	90-DEGREE PITCH MANEUVERS/ORBIT ADJUST MANEUVER AS REQUIRED
TBD	ASSUME MISSION MODE

NOTE: T = TIME OF LAUNCH.

Table 1-3. HCMM Orbit Adjust System Capabilities

ITEM	CAPABILITY (PER BURN)
MAXIMUM CHANGE IN ORBITAL VELOCITY	80 METERS PER SECOND
MINIMUM CHANGE IN ORBITAL VELOCITY	0.7 METERS PER SECOND
DIRECTION OF CHANGE IN ORBITAL VELOCITY	90-DEGREE PITCH MANEUVER REQUIRED IN THE DIRECTION DESIRED

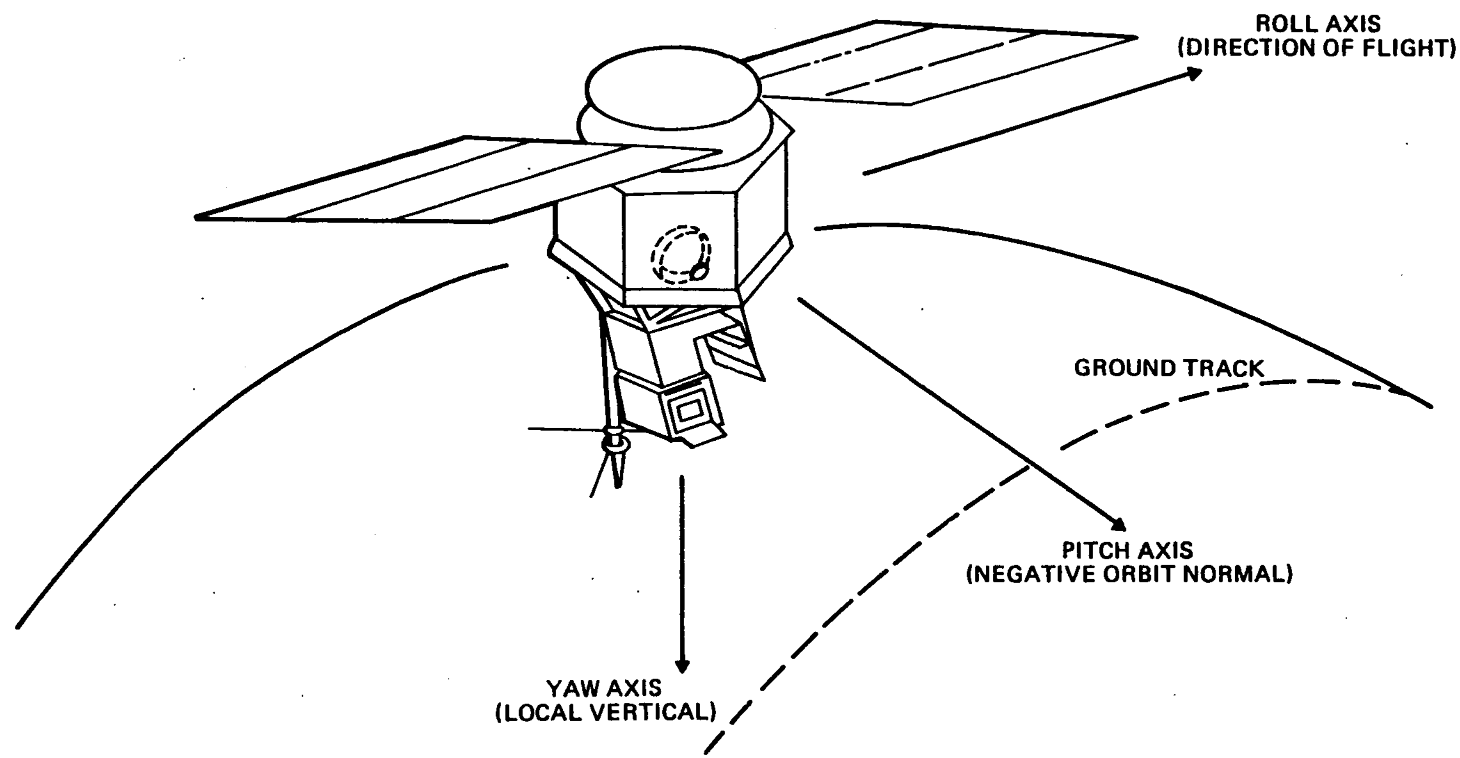


Figure 1-2. HCMM Orbital Coordinate System

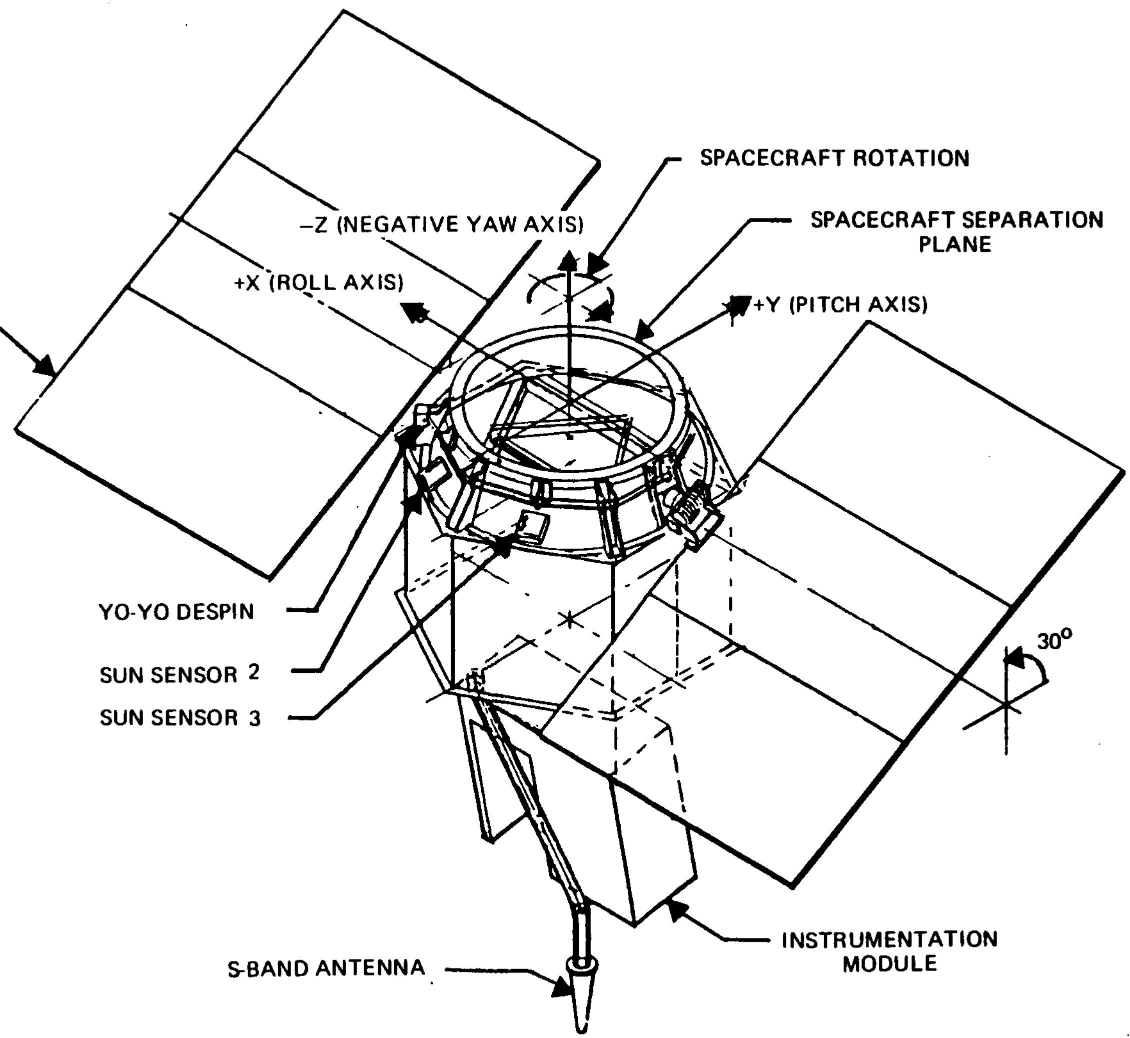


Figure 1-3. HCMM Spacecraft Body Coordinate System

spacecraft relative to the Earth's magnetic field, and the pitch rate approaches 2 revolutions per orbit. The pitch axis migrates toward the negative orbit normal as a result of magnetic torquing.

Once the pitch axis is within 10 degrees of negative orbit normal, on-orbit control laws are activated by ground command which spin up the momentum wheel to 1940 rpm and maintain the pitch angle near zero. Within a few orbits, the attitude and attitude rates stabilize to within the following limits:

- Pitch 0 ± 1 degree
- Roll 0 ± 1 degree
- Yaw 0 ± 2 degrees
- Rates, all three axes 0 ± 0.01 degree per second

1.1.4.2 Orbit Adjust Mode

Once a stable attitude has been attained, the orbit must be trimmed, or circularized. This is accomplished by commanding a pitch angle of ±90 degrees, thus placing the spacecraft Z-axis along the positive or negative velocity vector, so that the OAS, which is located on the spacecraft Z-axis, can apply appropriate thrusts. The thruster is fired for an appropriate length of time once per orbit to achieve circularization. This orientation is unfavorable for power supply maintenance, so after a maximum of 13 orbits of thrusting, the spacecraft is commanded back to zero pitch and the batteries are recharged. A minimum of five orbits is then required to recharge the batteries. The OAS maneuvers can be repeated as required, but should require no more than 2 weeks.

1.1.4.3 Mission Mode

After the final OAS maneuver, the spacecraft is in mission mode and should remain so for the remainder of the mission (1 year). If the roll attitude error from the infrared horizon sensor system exceeds 10 degrees, the control laws may be switched from on-orbit to acquisition mode. When an acceptable attitude has been reacquired, a ground command is required to resume the on-orbit control laws and the spacecraft returns to mission mode. Control of the wheel momentum is accomplished automatically onboard using magnetic torquing coils.

1.2 ATTITUDE DETERMINATION HARDWARE

This subsection describes the hardware used for determination of attitude, including the two-axis digital Sun sensors, the magnetometers, and the infrared Earth horizon scanner.

1.2.1 Sun Sensors

The Sun sensors to be flown on HCMM are Adcole model 16764. A complement of three sensors will provide continuous Sun coverage during daylight passes in the Sun-synchronous orbit. The sensors each provide two-axis Sun information, yielding a sunline direction relative to the sensor. Each sensor consists of two identical components, one for each Sun sensor axis, as depicted in Figure 1-4. Sunlight passes through the slit and is refracted onto the reticle pattern. The photocells beneath each slit of the reticle pattern produce an 8-bit encoded representation of the angle between the sensor Z-axis and the projection of the sunline on the sensor X-Z or Y-Z plane. The output from the most intensely illuminated of the three sensor heads is placed in telemetry; onboard determination of the most intensely illuminated sensor head is performed by examining the Automatic Threshold Adjust (ATA) photocell output (the largest of the three ATA outputs is selected).

The FOV of each Sun sensor axis is 128 degrees; hence, each sensor head provides a 128-by-128-degree FOV. The 8-bit reticle pattern yields an uncertainty of ±0.25 degree in the measured angles, which translates into a maximum Sun vector arc-length uncertainty of 0.35 degree.

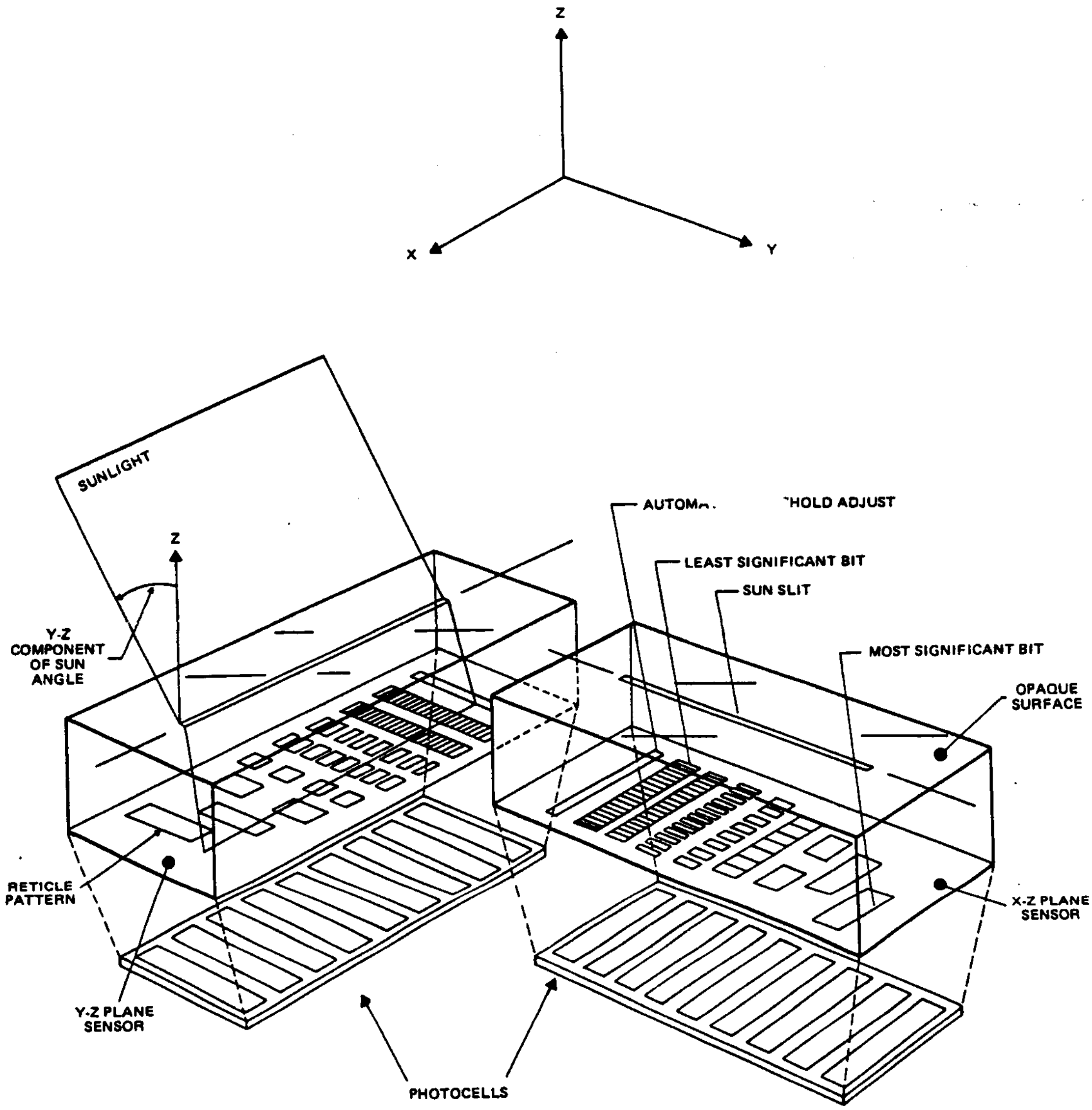


Figure 1-4. HCMM Two-Axis Digital Sun Sensors

The orientation of the sensor heads on HCMM is shown in Figure 1-5. The transformation from spacecraft to sensor coordinates is given by an ordered 3-1-3 Euler transformation using the following nominal angles:

- Sensor 1: $\theta_1 = 0$ degrees, $\theta_2 = 50$ degrees, $\theta_3 = 45$ degrees
- Sensor 2: $\theta_1 = 30$ degrees, $\theta_2 = 137$ degrees, $\theta_3 = -90$ degrees
- Sensor 3: $\theta_1 = -30$ degrees, $\theta_2 = 137$ degrees, $\theta_3 = -90$ degrees

The sensors will be aligned by Boeing Corporation to an estimated accuracy of 0.1 degree (each axis).

1.2.2 Magnetometers

Three Schoenstedt model fluxgate magnetometers are mounted parallel to the spacecraft X, Y, and Z axes. Output of the three magnetometers consists of the components of the Earth's magnetic field along the sensor axes with a full scale reading of approximately ±600 millioersteds. This output is used for

- Yaw attitude determination in conjunction with horizon sensor data
- Pitch, roll, and yaw attitude determination in conjunction with Sun sensor data
- Input to the onboard control electronics which differentiates the magnetometer data and then uses both the magnetometer and the magnetometer rate data for the acquisition and on-orbit control laws

The least significant bit corresponds to $1200/2^8 = 4.68$ millioersteds, or a maximum error of ±2.34 millioersteds (1.35 millioersteds, root mean square (rms)). The internal alignment of the sensor package is performed by the manufacturer to an accuracy of ±0.25 degree along each axis; external alignment is performed by the Boeing Corporation to an estimated accuracy of ±0.25 degree along each axis. The error budget is given in Table 1-4.

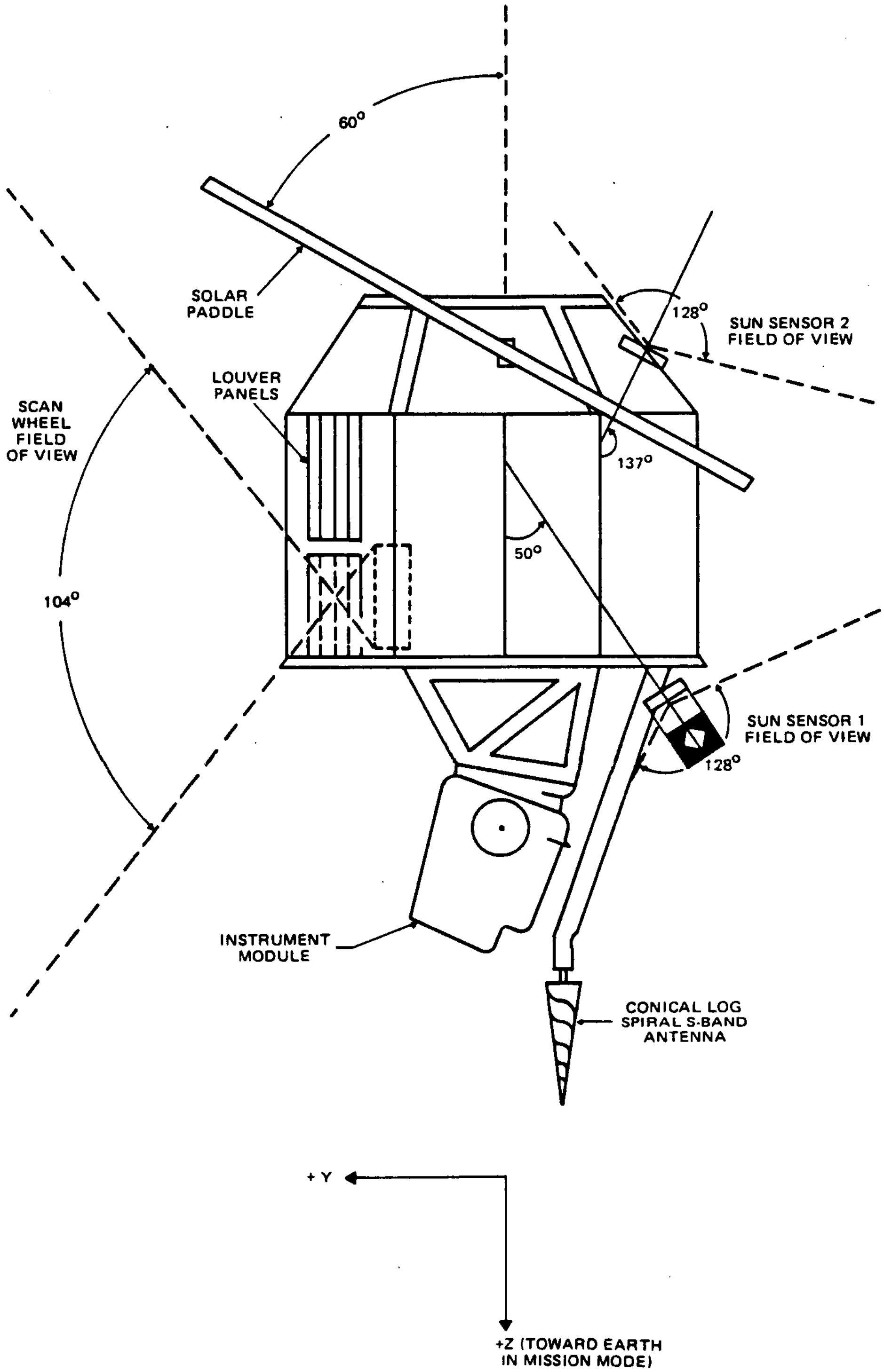


Figure 1-5. Location and Orientation of Sun Sensors on HCMM Spacecraft

Table 1-4. Magnetometer Error Analysis

ERROR SOURCE	MAGNITUDE	ERROR IN MILLIGAUSS (EACH AXIS)
INSTRUMENT NULL ERROR (IN ZERO FIELD)	± 7.0 MILLIVOLTS	1.7
INSTRUMENT SCALE FACTOR ERROR	1 PERCENT	1.0
TELEMETRY QUANTIZATION ERROR	± 10 MILLIVOLTS	2.3
TELEMETRY NOISE DRIFT	± 10 MILLIVOLTS	2.3
SPACECRAFT UNCOMPENSATED FIELD		5.0
ROOT-SUM-SQUARE (RSS) OF SINGLE-AXIS ERROR (IN MILLIGAUSS)		6.28

1.2.3 Infrared Earth Horizon Detector and Scanwheel System

The scanwheel[1] assembly consists of a variable speed momentum wheel, powered by a two-phase induction motor, and an integral infrared (IR) optical detection system. This assembly, mounted on the spacecraft Y (pitch)-axis, serves to provide the following:

- Angular momentum along the pitch axis for spacecraft stability
- Attitude control about the pitch axis by variations in the wheel rate
- Pitch and roll angle data for ground-based attitude determination and for onboard use by the attitude control electronics

As shown in Figure 1-6, the major components of the wheel subsystem are the motor rotor, the flywheel rim, the magnetic pickup, and the magnetic pickup slug. The motor rotor and magnetic pickup slug rotate with the flywheel rim at a nominal speed of 1940 rpm in mission mode. This provides the spacecraft stability, and the onboard controller uses a sensed pitch error to vary the wheel speed to provide pitch control. As the wheel rotates, the magnetic pickup senses

[1]Scanwheel is a registered trademark of Ithaco, Inc.

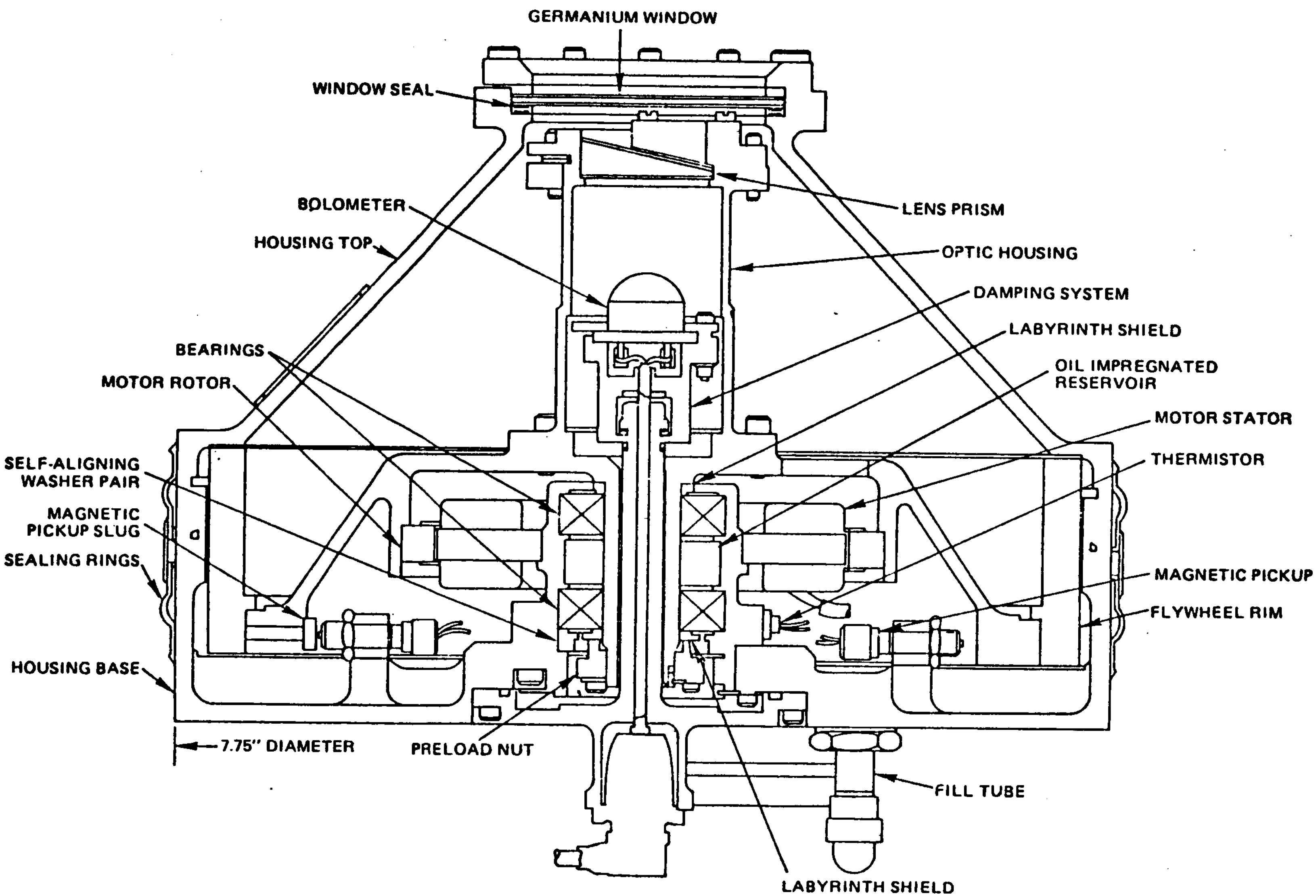


Figure 1-6. HCMM Scanwheel Cross Section

the passage of the magnetic pickup slug, thus determining the azimuthal position of the wheel in spacecraft coordinates at the time of the midpoint of the Earth scan.

The infrared optical subsystem consists of a hyperimmersed dual-flake thermistor (thermally sensitive resistor) bolometer, a rotating prism with a lens, and a germanium window that is coated with an interference filter to establish the 14- to 16-micron optical pass-band. The germanium window and bolometer are attached to the body of the spacecraft, while the lens and prism rotate with the wheel. Figure 1-7 indicates the passage of light through the optical system. For a fixed position of the lens, radiation near the 14- to 16-micron wavelength pass band which lies in the 2-degree-by-2-degree-square FOV passes through the window and lens and is focused on the bolometer. Note that the line of sight for each flake of the bolometer is slightly different. As the lens wedge rotates, the FOV sweeps out a 45-degree cone centered about the spin axis of the wheel. The response of the bolometer to the incoming light source and the resultant signal processing is shown in Figure 1-8. As the CO_2 layer of the Earth enters the FOV of the optical system, the radiation focused on each flake raises its temperature, thereby altering the resistance, and yielding an output signal proportional to the intensity of the incoming radiation. This signal is maximum when the FOV is entirely within the CO_2 layer, and decreases as the FOV enters and leaves this layer (Figure 1-8 (a) and (b)). The two times when the output signal attains a fixed value are used to form square waves (Figure 1-8 (c) and (d)), which represent the percentage of time that the CO_2 layer was in the FOV of each flake. For purposes of Sun rejection, a logical "and" is performed on these square waves to produce the resultant signal (Figure 1-8 (e)). The Sun rejection is accomplished by the offset FOVs for the two flakes, which ensures that the Sun cannot be sensed by both flakes simultaneously. The "anded" signal is split by the sensing of the magnetic pickup to produce two distinct signals,

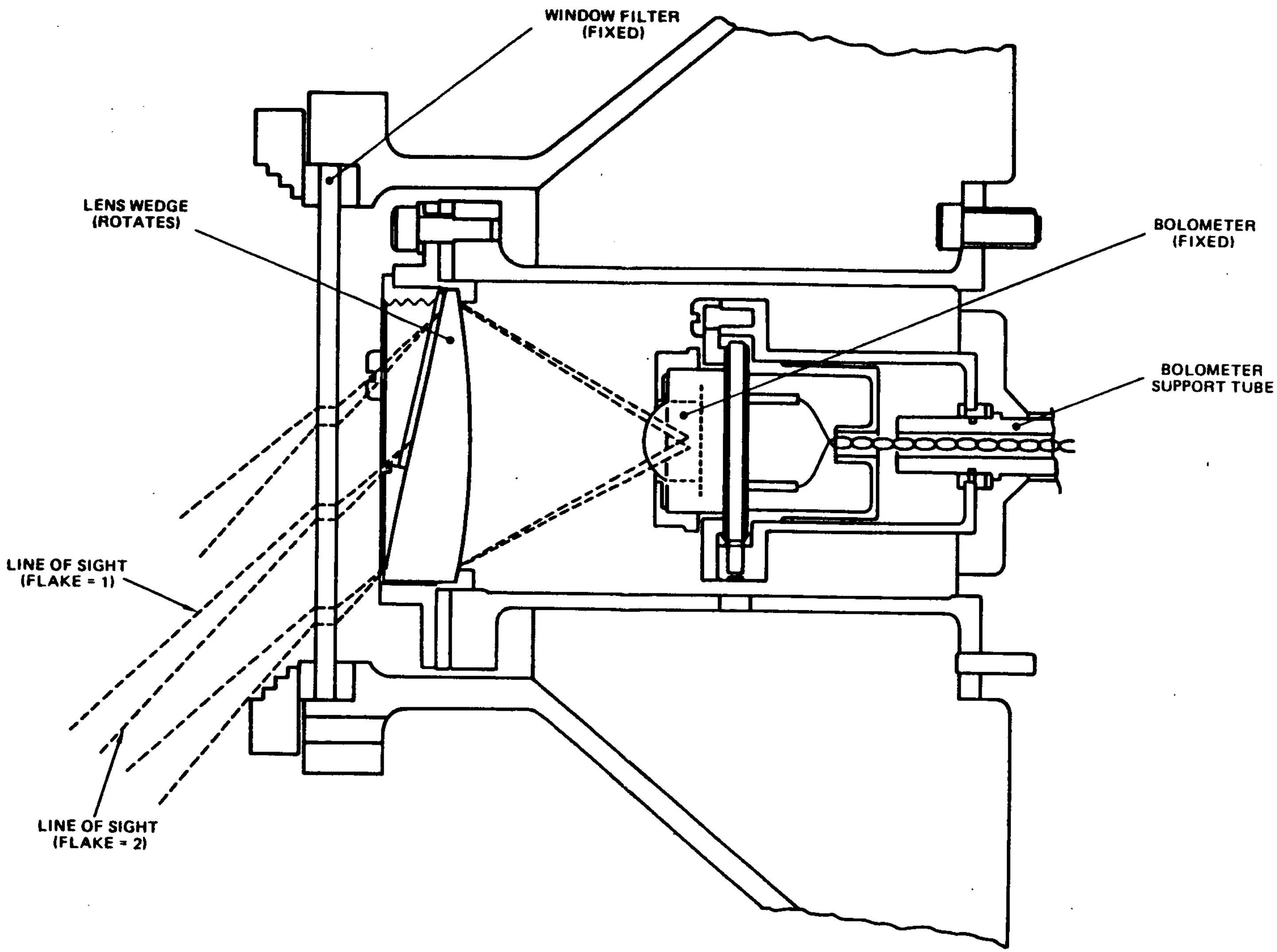


Figure 1-7. Conical Scanner Optics

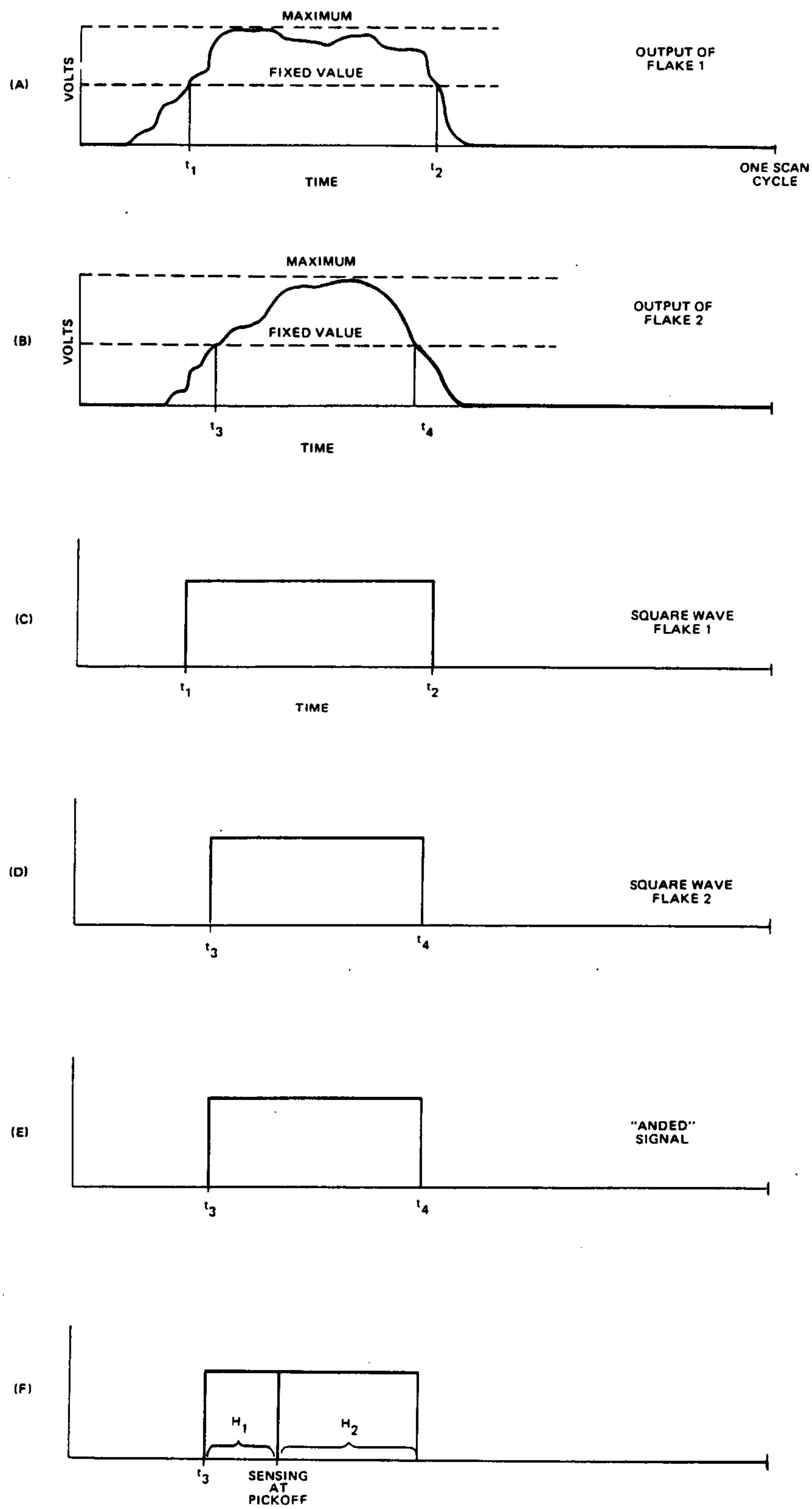


Figure 1-8. Infrared Scanner Signal Processing

H_1 and H_2 (Figure 1-8 (f)), which appear in the telemetry stream as the duty cycle data. The roll telemetry data is proportional to the sum of these two signals, and the pitch telemetry data is proportional to the difference of these signals. Table 1-5 lists the attitude error sources for the scanwheel system.

1.3 ATTITUDE CONTROL HARDWARE

The hardware used for attitude control of HCMM consists of the following:

- Magnetometers
- A scanwheel assembly
- A control electronics assembly (CEA)
- An electromagnet assembly (EMA)

The magnetometers and scanwheel assembly are described in Section 1.2; the remaining components are described in the following subsections.

1.3.1 Control Electronics Assembly

The CEA comprises the electronics necessary to process the output from the magnetometers during acquisition mode, and from both the magnetometers and the scanwheel in mission mode, and to implement the attitude control laws. It consists of a signal processor, attitude computer, power supply, scanwheel motor driver, magnetometer electronics, and associated magnetic control components.

The signal processor produces Earth pulses from the scanwheel as input to the attitude computer. The attitude computer then filters the Earth pulses with a lead-lag network and computes error signals for pitch and roll. The power supply operates from the 28-volt spacecraft power buss and generates most of the voltages required by the CEA, including the 650 hertz phasing and sawtooth voltage required by the motor driver. The motor driver contains a switching regulator and a bridge circuit to provide a linear (proportional to input signal)

Table 1-5. Scanwheel Error Analysis

ERROR SOURCE	PITCH (DEGREES)	ROLL (DEGREES)
ALIGNMENT ERROR BOLOMETER TO SCANWHEEL REFERENCE POINT (± 0.1 DEGREE); REFERENCE POINT TO MOUNTING HOLES (± 0.01 DEGREE)	± 0.15	± 0.15
RADIANCE EFFECTS ALTITUDE CHANGES (APPROXIMATELY ± 5.6 KILOMETERS)	± 0.10	± 0.15
AMBIENT TEMPERATURE	± 0.10	± 0.10
OBLATENESS EFFECT (± 10 KILOMETERS)	± 0.10	± 0.27
HORIZON LOCATION (RADIANCE PROFILE)	± 0.25	± 0.25
SIGNAL PROCESSING	± 0.03	± 0.03
ORBIT ECCENTRICITY (0.02 DEGREE PER KILOMETER)	—	± 0.10
ROOT SUM SQUARE (RSS)	± 0.34	± 0.45

torque to the rotating scanwheel. The magnetometer electronics contain electronic components required for processing data from the three fluxgate magnetometers. The magnetic control system provides spacecraft despin capability, roll and yaw acquisition and attitude control, and nutation damping.

1.3.2 Electromagnet Assembly

The EMA consists of three coils with ferromagnetic cores encapsulated in fiberglass. Each coil is 15-1/2 inches long and 3/4 inch in diameter. The three electromagnets are positioned on orthogonal axes coinciding with those of the three magnetometers. The EMA provides a maximum magnet strength of 10,000 pole-centimeters along all three axes; the instantaneous values are determined by control laws implemented by the CEA. The magnetic fields created by the EMA also bias the magnetometer data; this effect will be minimized by mounting the magnetometers on a boom. Linearity and hysteresis properties of the magnets are to be determined.

1.4 ONBOARD CONTROL SYSTEM

The onboard control system performs the following major functions:

- Acquisition of initial attitude after final spinup of the scanwheel
- Maintenance of nominal attitude throughout the mission

To accomplish these goals, an integrated system consisting of the component packages discussed in Section 1.3 is provided. A brief discussion of the types of control provided by these packages and the phases of the mission in which they are used follows.

Four types of control are provided for acquisition and maintenance of spacecraft attitude. They are (1) attitude acquisition, (2) pitch control, (3) roll, yaw, and nutation control, and (4) scanwheel desaturation.

1.4.1 Attitude Acquisition

During the attitude acquisition phase of the mission, the spacecraft attitude will be controlled by magnetic torques created by the EMA. The instantaneous values of these torques are determined by a feedback loop consisting of the magnetometers and the CEA. The CEA attempts to minimize the time derivative of the value of the Earth's magnetic field in the spacecraft coordinate system.

The consequences of these actions are:

1. The pitch or wheel axis is driven to within 10 degrees of negative orbit normal.
2. The spacecraft tracks the Earth's magnetic field, i.e., the rate about the pitch axis approaches a mean of 2 revolutions per orbit inertially, or 1 revolution per orbit relative to the Earth.

1.4.2 Pitch Control

In addition to the three-axis control afforded by the EMA, control about individual axes is provided by separate control laws driven by information from the scanwheel. Control about the pitch axis is provided by a feedback loop consisting of the scanwheel and the CEA. As mentioned in Section 1.2.3, pitch control is accomplished by measuring the time lapse between the mid-Earth scan time and the detection of the magnetic pickup in the body. The pitch angle is given by the time difference multiplied by the angular velocity of the wheel. The wheel speed is varied by the CEA to maintain pitch within 1 degree of zero.

1.4.3 Roll, Yaw, and Nutation Control

The scanwheel output and the CEA feedback loop are also used for control of roll and, indirectly, yaw. The Earth-in and Earth-out times provide a measure of the fraction of the scanwheel revolution which detects the Earth. For a given altitude and for zero roll this value is known, and a deviation therefrom indicates a nonnominal roll angle. Corrections to this angle are made automatically via the EMA. Through quarter-orbit coupling this will correct, on a time average, both roll and yaw. Nutation damping is accomplished by modulating the Y-axis electromagnet dipole strength 180 degrees out of phase with the Y-axis magnetometer rate.

1.4.4 Scanwheel Desaturation

The scanner wheel speed is continuously monitored by the attitude control electronics assembly. If the wheel speed differs by more than 40 rpm from the nominal value of 1940 rpm, electromagnet commands are issued to generate magnetic torques which alter the total spacecraft angular momentum and drive the wheel speed back to the nominal value.

SECTION 2 - ATTITUDE DETERMINATION AND CONTROL SECTION REQUIREMENTS

2.1 INTRODUCTION

The AEM is a program designed to collect scientific data using a similar base module for each mission. In addition, off-the-shelf hardware will be used whenever possible. This common base module concept was designed to collect scientific data in a very cost-effective way. The HCMM is the first AEM. This vehicle is scheduled to be launched in April 1978 from the Western Test Range (WTR) on a Scout booster. Nominally the Scout launch vehicle will inject the spacecraft into a 600-kilometer circular orbit with an Earth equatorial inclination of 97.79 degrees. This will result in a Sun-synchronous orbit. Time of launch will be chosen such that the spacecraft will have a 2:00 p.m. (±1 hour) ascending node. This Earth orbit will allow the scientific package to thermally map the Earth at a constant time of day throughout the life of the mission. The HCMM spacecraft will have an OAS to correct possible in-plane dispersion errors caused by a nonnominal orbital state vector at time of orbit injection. This propulsion system has the capability of correcting eccentricity and orbital height, within certain limits (70 meters per second), but will not be used to correct out-of-plane orbital errors, i.e., equatorial inclination. The orbit adjust system uses hydrazine as a propellant.

Let TL represent time of launch. Then, the nominal sequence of events is as follows:

Time	Event
TL + 620 seconds	Fourth stage burnout
TL + 913 seconds	Spacecraft separation
TL + 923 seconds	Deploy yo-yos
TL + 965 seconds	Turn Attitude Control System (ACS) on Acquisition Mode

Time	Event
TL + 975 seconds	Wheel to 800 rpm
Fourth Orbit	Command on-orbit mode
Eighth Orbit	Acquisition completed
TL + 1.25 days:	Start orbit trim maneuvers
TL + 16.25 days:	Orbit trim completed; start operations mode
TL + 365 days	End of mission

The nozzle of the OAS will nominally be pointed away from the Earth along the Earth to spacecraft vector and will thus have to be pitched ±90.0 degrees, depending on whether velocity is to be added or subtracted. The spacecraft will stay in this propulsion system firing attitude for a maximum of 13 orbits and then will be commanded to the nominal 0.0-degree pitch attitude for at least 5 orbits so that the spacecraft batteries can be recharged. The above ±90.0-degree pitch maneuver is initiated by a telemetry command, with the autonomous attitude control system onboard the spacecraft performing the maneuver.

2.2 SPACECRAFT HARDWARE

The HCMM will have a single infrared (IR) scanwheel assembly (SWA-B) with dual flake bolometer optics, one Control Electronics Assembly (CEA), a magnetometer triad, and three Electromagnet Assemblies (EMA). The SWA is a reaction wheel scanner with a variable speed momentum wheel, powered by a two-phase induction motor. It contains an integral IR optical system with an optical passband of 14 to 16 microns. The optical attitude IR sensor has a square field of view of 2.0 degrees on each side. The momentum vector associated with the scanwheel is along the spacecraft negative pitch axis which is nominally orthogonal to the spacecraft's orbital plane. The IR scanner optical system generates a 45-degree cone about the spacecraft pitch axis. The CEA package controls the scanwheel, processes the signals, and implements the system control laws. The orthogonally mounted electromagnets provide control torques for acquisition, roll/yaw control, desaturation, and nutation damping. The 10,000 pole-centimeter units are 1 inch in diameter, 12 inches long, and contain a ferromagnetic core. The IR SWA-B, the CEA, and the electromagnets are all built by the Ithaco Corporation for HCMM. A Schoenstedt triaxial flux-gate magnetometer similar to models flown on Orbiting Solar Observatory (OSO) and the Geodynamics Experimental Ocean Satellite (GEOS) measures the three components of the local magnetic field vector. This data is processed by CEA to generate coil current commands. It is also telemetered to ground for attitude determination. Three Adcole Sun sensor heads, each having a field of view of ±64 degrees in two axes with an accuracy of ±0.25 degree, provides attitude determination data which is telemetered to ground for attitude determination but is not used in the onboard ACS. If data is available from one sensor, it is sufficient to provide yaw solutions in the nominal orbit. Three heads are required to accommodate nodal drift and orbital insertion errors.

2.3 OPERATIONAL REQUIREMENTS

It is to be understood that operational requirements refer to those functions that the Attitude Determination and Control Section (ADCS) (Code 581.2) will assume from launch through attitude capture and orbit adjust phases of the mission. During the attitude capture phase of the mission, the following telemetry values will be monitored:

1. Magnetometer output
2. Sun sensor output (when possible)
3. Roll and pitch residuals from the autonomous control system
4. Electromagnet output
5. Scanwheel speed
6. Scanwheel duty cycle

In addition, the following status telemetry will be monitored:

1. Acquisition mode on/off
2. Sun sensor on/off
3. Pitch maneuver yes/no
4. Magnetometer on/off
5. Magnetometer in/out of control loop

An attempt to time tag the following events should be made:

1. Magnetic capture of spacecraft attitude, i.e., when the spacecraft starts to track the Earth's magnetic field

2. Coarse attitude capture, i.e., when the spacecraft pitch and roll angles are between ±5.0 degrees

3. Fine attitude capture, i.e., when the spacecraft pitch and roll angles are between ±2.0 degrees

4. On-orbit mode is achieved, i.e., when the autonomous control system is maintaining attitude within ±2.0 degrees in yaw, ±1.0 degree in pitch and roll

During the orbit adjust phase of the mission, attitude determination will be performed at each station pass. The telemetry output previously mentioned will be monitored.

When the ±90.0-degree pitch command is executed, the wheel speed, IR sensor output, magnetometer output, and possibly Sun sensor output will be monitored to verify successful completion of the maneuver. After an orbit adjust propulsion system firing, the attitude will be continuously monitored at every available pass and a determination made as to damping of motion induced by the orbit adjustment maneuver. The above activities will continue until the orbit adjust phase is completed. Also during the orbit adjust phase, an in-flight data analysis activity will be initiated to extract roll and residual dipole biases.

This activity will require the use of an auxiliary bias determination software program, a low-level analysis support, and low-level operational support for approximately 2 weeks after the orbit adjust phase. This activity will improve the performance of the autonomous attitude control system as well as the quality of definitive attitude information produced.

2.4 DEFINITIVE REQUIREMENTS

The HCMM definitive requirements include processing the IR scanner data, the Sun sensor data, and the magnetometer data received over the data link from the Information Processing Division (IPD) and computing definitive attitude and body angular rates for the lifetime of the mission (1 year). HCMM has no tape recorders; therefore, no instrument data or definitive attitude data will be available except over a tracking station. Data will be taken from the spacecraft approximately once per orbit. The definitive attitude requirements are yaw ±2.0 degrees, pitch ±0.5 degree, and roll ±0.7 degree. Let (X', Y', Z') represent the spacecraft body coordinate system where X' is the roll axis, Y' is the pitch axis, and Z' is the yaw axis. Let (X, Y, Z) represent the orbital coordinate system where $\bar{Z} = -\bar{R}$ (spacecraft Earth-centered position vector), $\hat{Y} = (\bar{V} \times \bar{R}) / |\bar{V} \times \bar{R}|$ where $\bar{V}$ is the spacecraft velocity vector, and then $\hat{X} = \hat{Y} \times \hat{Z}$. If K is the 2-1-3 Euler transformation, then

$$\begin{pmatrix} X' \\ Y' \\ Z' \end{pmatrix} = [K] \begin{pmatrix} X \\ Y \\ Z \end{pmatrix}$$

Attitude for output purposes, i.e., the values generated for the experimenters, will be defined by $[K]^{-1}$. Definitive attitude will be sent to the UNIVAC 1108 computer over the data link no later than 3 weeks after the ADCS receives the definitive orbit tape. The Multisatellite Attitude Determination System (MSAD) will be capable of providing IPD a tape of definitive results (backup). In addition, the telemetry processor will be capable of accepting tape input from IPD and the Multisatellite Operations Control Center (MSOCC). Memoranda of understanding from the ACA to both IPD and MSOCC concerning telemetry data transmission to ADCS and definitive attitude data transmission to IPD are provided in Appendix B.

2.5 DEFINITIVE AND NEAR-REAL-TIME SOFTWARE CONSIDERATIONS

System characteristics for MSAD/HCMM include the following:

- It will operate in less than 350K bytes of core.
- It will use orbit ephemeris data in either the EPHEM format or an internal generator.
- The definitive attitude determination system will be capable of supporting near-real-time monitoring functions as well as definitive attitude determination functions.
- MSAD/HCMM will operate under the Graphic Executive Support System (GESS) on the IBM 2250s or Data Disc 6600s.
- MSAD/HCMM will write a general attitude file.

The system will consist of a driver to attach a telemetry processor, a data adjustment function, an attitude determination function that includes the necessary quality control functions, and a monitor function to support near-real-time monitoring activities.

Specific considerations for the MSAD/HCMM systems are the following:

- Telemetry Processor Driver
  - Handles data from data link or tape
  - Handles a 312-byte format in support of control center activity and a 3492-byte format from IPD for definitive attitude computations
- Data Adjustment
  - Adds biases to data
  - Adjusts times of observations
  - Smoothes observations as necessary
  - Formats data for further processing
- Attitude Determination
  - Determines attitude using deterministic and recursive techniques as on GEOS-C
  - Uses all available data sources (IR scanner, magnetometers, and Sun sensors, if applicable) to provide quality control checks
- Monitoring Functions
  - Monitors the acquisition phase of the mission
  - Monitors maneuvers in near-real time
  - Displays predicted and observed Sun angles, Earth width, spin rates, and attitudes

The MSAD/HCMM will be compatible with GESS, will provide display capabilities to control subsystem flow, permit data rejection, and display results. The system will be capable of the following:

- Operating in near-real time and definitive processing mode
- Processing attitude data received from MSOCC and IPD (TELOPS) via data link and backup magnetic tapes
- Processing standard 312-byte records
- Processing standard TELOPS records
- Data smoothing, validation, time checking, and filtering (if necessary)
- Displaying and plotting the attitude telemetry data on a cathode ray tube (CRT)
- Converting data to engineering units
- Providing automatic data logging functions
- Writing a bias determination data file

The MSAD/HCMM attitude system will be capable of smoothing the attitude results and generating a definitive attitude data set for transmission to IPD. A separate specification shall be developed which will define a technique which can be implemented in MSOCC to provide pitch, roll, and yaw computations for display in mission mode. The IR and Sun sensor shall be used to provide this support.

2.6 SIMULATOR REQUIREMENTS

An attitude simulator should be provided to create realistic attitude data to verify the functions and computations of the attitude determination routines.

The attitude simulator should be capable of the following:

- Operating in less than 200K bytes of core
- Producing simulated Earth, Sun, and magnetic field data
- Creating a bias determination data set
- Simulating the various logical modes of operation of the spacecraft attitude sensors
- Operating in batch mode
- Creating a sequential data set for both 312- and 3492-byte formats
- Simulating anomalous sensor behavior and data dropout
- Adding biases to Sun, Earth, and magnetic field sensor data
- Adding white noise to observations
- Inserting gross errors at defined intervals to check rejection criteria
- Providing the capability to insert poorly time-tagged data to check time rejection criteria
- Providing for the creation of backup telemetry tapes

2.7 OTHER REQUIREMENTS

Additional software packages are required to provide the following capabilities:

- Quality control of definitive attitude results
- Analysis of in-flight data to determine roll bias for commanding and data processing purposes
- Analysis of in-flight data to determine residual magnetic dipole biases for commanding and data processing purposes

2.8 SCHEDULES

The following schedule is planned for the HCMM:

Time	Event
December 31, 1976	Attitude system functional specification document complete
February 15, 1977	Complete simulator design
April 1, 1977	Complete attitude software system design
April 1, 1977	Complete utility programs design
July 15, 1977	Simulator complete
October 1, 1977	Utility programs complete
December 1, 1977	Attitude software system complete
December 1, 1977	Begin acceptance testing
February 1, 1978	Complete acceptance testing

SECTION 3 - SUPPORT SYSTEM REQUIREMENTS

The requirements placed on the attitude support system during the attitude acquisition, orbit adjust, and on-orbit modes of the mission are outlined in Section 3.1. Section 3.2 summarizes the capabilities of the HCMM Attitude Support System, and Section 3.3 describes the computer environment necessary to achieve these capabilities. Section 3.4 discusses operational consideration during the various phases of attitude support for the mission.

3.1 SUPPORT REQUIREMENTS

3.1.1 Attitude Acquisition Mode Requirements

Launch of the HCMM spacecraft is scheduled for April 1978 at Vandenburg Air Force Base onboard a Scout Launch vehicle. Approximately 15 minutes after launch, the spacecraft will separate from the fourth stage of the launch vehicle with an imparted spin rate of approximately 150 rpm about the spacecraft Z-axis. At separation, an automatic sequence is initiated such that 10 seconds later a yo-yo despin device will be deployed to reduce the angular rate to approximately 2 rpm; 32 seconds later the attitude control hardware will be enabled, the acquisition control laws invoked, and the pitch momentum wheel spun to 800 rpm. For the remainder of the first four orbits, the acquisition control laws will torque the spacecraft pitch axis toward the negative orbit (the nominal mission mode attitude), and the inertial attitude rate about the pitch axis will approach 2 revolutions per orbit (rpo). After these two events have been accomplished, on-orbit control mode may be commanded.

The attitude support system will provide near-real-time monitor assistance during these events (see Section 3.4.1 for a description of near-real-time data processing). Because there are no tape recorders on the spacecraft, this assistance is provided only during times of station contact. The support system will verify that yo-yo despin, activation of the acquisition control laws, and momentum wheel spinup have occurred. The support system will also provide attitude and attitude rate information when there is sufficient sensor data. In addition, it will monitor the movement of the pitch axis toward negative orbit normal and determine when the spacecraft attitude is within mission constraints so the on-orbit control mode may be commanded on from GSFC.

3.1.2 Orbit Adjust Mode Requirements

The on-orbit control mode command activates the pitch control law, the roll/yaw and nutation control law, and the wheel desaturation control law. Following these attitude maneuvers, the spacecraft will achieve the mission mode attitude (pitch and roll angles within ±1.0 degree; yaw angle within ±2.0 degrees; all body rates less than 0.01 degree per second) using these control laws. In addition, the batteries will be charged in preparation for the orbit adjust maneuvers which will begin at approximately the 16th orbit.

The orbit adjust maneuver will be initiated by commanding a 90-degree pitch rotation from GSFC. The spacecraft will remain at the ± 90-degree pitch attitude (Z-axis in the desired thrust direction), and engine firings are commanded from GSFC approximately once per orbit for about 13 orbits. Following the orbit adjust maneuver, the pitch angle is commanded to zero and the spacecraft is returned to the autonomous control mode while the batteries recharge. This configuration is maintained for at least five orbits, after which the orbit adjust sequence is repeated as necessary. This 18-orbit cycle will continue until the orbit is circularized, which should occur within 15 days.

The attitude support system will provide a near-real-time attitude monitor capability. It will enable the analyst to determine that the mission mode attitude has been achieved prior to orbit adjust maneuvers. It will verify that the 90-degree pitch maneuver has been successfully accomplished and that the resulting roll, yaw, and wheel speed disturbances induced by the attitude maneuver have been controlled. After an engine firing, the analyst will confirm that the attitude disturbances induced by the thrust have damped before the next firing takes place. After an orbit adjust sequence has been completed, the analyst will monitor the maneuver to pitch zero and maintenance of the mission mode attitude. Near-real-time support will be provided throughout the orbit adjust phase.

3.1.3 Mission Mode Requirements

After the orbit adjust phase has been completed, the spacecraft enters the mission phase, during which nominal spacecraft operation and experimental data collection will occur. The four attitude support requirements for mission mode, which is scheduled to continue for 1 year, are as follows:

- Determination of the spacecraft residual dipole compensation
- Determination of the roll bias compensation
- Determination of scanner pitch and roll biases
- Determination of definitive attitude

If an uncompensated spacecraft residual dipole is present, it will interact with the Earth's magnetic field to create a disturbance torque. Large uncompensated dipoles could prevent the attitude control system from maintaining the mission mode attitude. Significant uncompensated dipoles will degrade the control system performance by causing sinusoidal roll and yaw oscillations and large angular momentum changes. The residual dipole compensation command will permit the biasing of the null value for each electromagnet dipole and will negate the effect of the residual dipole. The attitude support system will be used to determine the value of a residual dipole. The appropriate command will be uplinked in 80-pole-centimeter increments. Compensation is required to an accuracy of 400 pole-centimeters.

The precession control law applies a restoring magnetic torque which is proportional to the roll error signal, which is the difference between the observed roll voltage and a nominal value. This nominal value is commandable and should equal the voltage observed by the scanner at zero roll angle for the HCMM altitude. A value can be computed before lanunch for the expected spacecraft altitude and horizon sensor characteristics, but may require updating after launch to reflect the true altitude and sensor characteristics. The attitude support system will determine the value of the roll voltage compensation to permit the appropriate command to be uplinked.

Wheel mounted horizon scanners experience a variety of biases, including mounting misalignments, bolometer offsets, and electronic anomalies. Because HCMM is an Earth-oriented spacecraft in a nearly circular orbit, the geometry of the Earth relative to the scanner will change only slightly and therefore these biases will be manifested as nearly constant angular offsets on the pitch and roll data. The attitude determination support system will determine these offsets and incorporate them in the scanner data processing algorithms.

The attitude support system will determine definitive attitude solutions for all available data. These solutions, which consist of pitch, roll, and yaw angles, will be determined with an uncertainty (3σ) of 0.5 degree in pitch, 0.7 degree in roll, and 2.0 degrees in yaw.

3.2 SUMMARY OF SUPPORT SYSTEM CAPABILITIES

The support system specified to meet the HCMM attitude determination requirements consists of the HCMM Attitude Determination System (ADS), a data simulator, a scanner bias determination utility, a residual dipole bias determination utility, and a log interrogation and update utility.

3.2.1 HCMM ADS Overview

The HCMM ADS will operate under control of the Graphic Executive Support System (GESS), and will provide graphic and interactive capabilities as well as dynamic allocation of arrays and program interrupt interception. Telemetry processor, data adjustment, attitude determination, and output and logging functions are addressed.

3.2.1.1 Telemetry Processor Functions

The telemetry processor functions include reading, unpacking, and converting data from a tape or disk telemetry data set. The telemetry data set will be either a standard attitude data link (ADL) data set containing data processed by MSOCC or IPD. The data is processed in batch mode. Quality checking of the data is performed if requested, including self-consistency of the ground attached time and spacecraft clock.

The basic sensor data types are pitch and roll voltages and duty cycle from the infrared (IR) horizon scanner, two digital Sun angles and a sensor head identifier from the Sun sensors, and three voltages from the magnetometers. When Sun data are not available, flag values are provided. Data relating to the onboard control system performance, scanwheel speed, and electromagnet data are also unpacked and, if requested, written to the electromagnet data set and the wheel speed data set which will be used for subsequent determination of the residual dipole bias.

Additional data types such as magnetometer rate data and status flags from the control system are unpacked, as required, for initial evaluation of the control system performance.

3.2.1.2 Data Adjustment Functions

The data adjustment functions include reading data from the magnetometer and scanner biases from data sets, rejecting outlying data, and reducing the data volume.

Spacecraft ephemeris is generated either from external data sets or from an internal orbit generator, and inertial Sun and geomagnetic field vectors are calculated. These vectors are used for validation of sensor data. The IR scanner duty cycle data is converted to roll angle data in a manner which compensates for errors caused by Earth oblateness and horizon radiance variations.

IR scanner, magnetometer, and Sun sensor data are used to construct the Earth-to-spacecraft, magnetic field, and Sun unit vectors in spacecraft coordinates.

The validation module performs three basic types of data checks. These are attitude independent checks, null attitude checks, and smoothing. Attitude independent checks are applied to a single data type, such as a comparison of the magnitude of the magnetic field vector in inertial and spacecraft coordinates, and multiple data type checks, such as a comparison of the angle between the Sun and magnetic field vectors in inertial and spacecraft coordinates. These attitude independent checks are made on all available data in both acquisition and on-orbit modes. Null attitude checks compare sensor data to predicted values based on known ephemeris information and the null attitude (pitch, roll, and yaw zero). This check is only valid during mission mode when the spacecraft will be maintained near the null attitude by the onboard control system. Smoothing of the data for validation purposes can be performed in either acquisition or on-orbit mode.

The data volume is reduced from the measurement frequency (approximately 1 second^{-1}) to the attitude solution frequency (approximately 0.1 second^{-1}) by either preaveraging the data or smoothing. Various quality assurance parameters, which give quantitative description of the data, are also computed. Data needed to determine scanner biases is optionally written to the scanner bias utility source data set.

3.2.1.3 Attitude Determination Functions

The primary attitude determination function is determining a three-axis attitude expressed as pitch, roll, and yaw angles. In acquisition mode, the attitude is computed deterministically from Sun sensor and magnetometer data when both data types are available (IR scanner data are not available in acquisition mode). In addition, pitch, roll, and yaw rate information is determined from the attitude solutions. Both attitude and attitude rates are then displayed for monitoring purposes.

In on-orbit mode, attitude solutions are computed either deterministically (point by point) or by use of a differential corrector which employs a time-dependent polynomial model for attitude propagation. If deterministic solutions are desired, pitch, roll, and yaw angles are computed from IR/Sun, IR/magnetometer, and Sun/magnetometer data combinations when such data are available. Error statistics and weights, which are based on the intrinsic sensor data accuracy and reference vector geometry, are used to obtain average pitch, roll, and yaw angles. These results, together with the error statistics, are smoothed, if desired.

When deterministic solutions are degraded because of insufficient or poor quality sensor data or because of large sensor biases, a differential corrector is used to compute attitude results and magnetometer biases. The state vector consists of the coefficients of a polynomial approximation to the attitude time dependence and biases on each magnetometer. The differential corrector estimates the values of the elements of the state vector and determines the uncertainty in this estimation. The polynomial coefficients are used to evaluate the attitude at requested times in the processing interval.

In the definitive mode, magnetometer biases are optionally determined from IR scanner and Sun sensor data for daylight passes and written to the magnetometer biases data set.

3.2.1.4 Output and Logging Functions

The ADCS requires that procedures be established to ensure the quality of attitude solutions delivered to IPD for the experimenter. To satisfy these requirements, a direct access log data set will be maintained containing a summary of all data processed. This data set serves the following three functions:

- Verifying the quality of the data recently processed (perhaps on a daily basis) prior to sending results to IPD. If the data quality is unacceptable, a determination of the problem must be made so that the data may be reprocessed.
- Maintaining a summary log of the attitude data sent to IPD.
- Logging the status of data received from IPD and producing hard-copy deliverables as specified by the ADCS.

The log data set, which is described in detail in Section 5.6.7, contains information relating to the following general areas:

- Header information (start and stop time of the data, date, etc.)
- Quality of the data (amount of data flagged by the telemetry processor and data adjustment functions)
- Quality of the attitude solutions
- Current disposition of the data
- Comments of operators and analysts regarding the data

The output and logging functions include writing a summary as described above to the log data set after each segment of data has been processed. It provides the user with a choice of overwriting an existing summary or concatenating the current summary to existing log data. The default will be to add the summary to the existing data until the log data set is full. Attempts to add data to a full log will result in a message to the operator in interactive mode. If the operator wishes to proceed, or if the program is executing in the noninteractive mode, the data in the log data set is scrolled, deleting the oldest summary in the data set and adding the current summary. The deleted summary will be retained in hardcopy output.

The attitude solutions will also be written to the definitive attitude history file. This data includes the following:

- A summary of the quality of the solutions for the segment
- Pitch, roll, and yaw angles at specified time intervals (nominally every 10 seconds)
- The GMT associated with each set of values

3.2.2 Data Simulator

The data simulator provides data similar in both form and content to that produced by the HCMM attitude determination and control hardware. This data will be used to test the HCMM ground support software, to train HCMM ground support attitude determination and control personnel, and to evaluate the performance of the onboard control system. The simulator, which is executed in batch mode, simulates all spacecraft control modes: acquisition, orbit adjust, and mission mode.

The simulator input consists of the initial attitude and attitude rate, spacecraft ephemeris, spacecraft dynamics parameters, sensor hardware parameters, systematic and random error control parameters, and integrator control parameters. The input attitude and attitude rate are converted to the spacecraft coordinate system to integrate the equations of motion using an Adams-Bashforth integrator. Gravity-gradient and solar radiation pressure environmental disturbance torques are included. Control system torques are computed from models of the onboard control system and the electromagnet and scanwheel hardware. Hardware models of the Schoenstedt three-axis fluxgate magnetometer, Adcole two-axis digital Sun sensors, and Ithaco scanwheel, which include misalignments and biases, are combined with ephemeris data to simulate analog sensor output at requested time intervals. Analog data from the attitude sensors and attitude control hardware are then digitized to simulate the OCC and/or IPD output format. The digitization process includes the capability to alter the data to reflect the systematic and random errors which may occur in telemetry data. Finally, the telemetry data may be written to data sets to be used in the evaluation of the ground support software. In addition, analytical data, including a truth model, may be written to data sets which will be used to exercise the attitude determination system during development. Appropriate truth model data are plotted.

3.2.3 Scanner Bias Determination Utility

The scanner bias determination utility provides the capability to determine the constant pitch and roll biases on the IR scanner data and to make these biases available to the attitude determination system processing algorithms. The utility, which is executed graphically and interactively under GESS, is a batch differential corrector which uses IR scanner, Sun sensor, and ephemeris data to estimate the scanner biases.

Input data, which consists of IR scanner pitch and roll angle data, Sun sensor data, and ephemeris information, is read from the scanner bias utility source data set. The observation, which is the third component of the Sun unit vector in orbital coordinates, is calculated for each time, and the observation model, which utilizes IR scanner data, Sun sensor data, and the initial estimate of the biases, is computed. The differential corrector is then applied, and an estimate of the biases, the uncertainties of the estimates, and the mean and standard deviations of the residuals are computed. These estimates are then used to calculate new values for the observation model, and the differential corrector is reapplied to obtain new estimates of the biases. This cycle is repeated until the uncertainties of the estimates and the mean and standard deviations of the residuals are within a specified tolerance or until a maximum number of iterations have been reached. The final estimates of the pitch and roll biases may then be written to the scanner biases data set provided that the uncertainties of the estimates and the mean and standard deviations of the residuals are sufficiently small. A hardcopy printout of all results is provided.

3.2.4 Residual Dipole Bias Determination Utility

The residual dipole bias determination for HCMM will be accomplished with the ADGEN/HCMM program in an interactive graphics mode. This program consists of a batch differential corrector which is designed to solve for residual dipole biases and, optionally, initial values of the attitude angles and rates. The observations will be pitch, roll, and yaw angles from the HCMM attitude history file. Other input data to this utility include the electromagnet and scanwheel speed data from the automatic control system, spacecraft ephemeris data, and the input state in terms of body rates and pitch, roll, and yaw angles. Several passes of data will be processed in a batch mode to determine the spacecraft residual dipoles.

The system accepts spacecraft configuration input, such as sensor and wheel orientations and body moments of inertia, in a spacecraft reference system. All quantities are then transformed to the body principal coordinate system for the integration of the equations of motion. Environmental torque models include aerodynamic drag, solar radiation pressure, magnetic dipole, and gravity-gradient torques. In addition, there is a provision for a constant torque in the spacecraft body coordinates which can be used to compensate for model inadequacies. The equations of motion are integrated using either a fourth order Runge-Kutta method or an Adams-Bashforth-Moulton predictor-corrector of 2nd to 10th order. The differential corrector converges if the change in each solve-for variable is less than a tolerance value specified for that variable.

When the residual biases have been determined, a hardcopy printout of the results is provided. If the values of the estimated residual biases are sufficiently large to suggest that control system performance will be significantly degraded, and if the uncertainties in these estimates are sufficiently small, the electromagnet coil biases will be commanded to negate the effect of these residual magnetic dipole biases.

3.2.5 Log Interrogation and Update Utility

The log interrogation and update utility is used both for quality assurance and for loading the transmission data set.

This utility is used by analysts to examine periodically the quality of attitude solutions for each pass of data. This examination is accomplished interactively and in batch mode. Based on the quality of the solutions as summarized in the log data set, the analyst sets a flag in the log summary of each pass of data indicating whether or not the data should be transmitted to the IPD.

On a daily basis, operations personnel at GSFC interrogate the log data set to determine which passes of data, if any, are ready for transmission to the IPD. The interrogation and update utility allows the operator to perform this inspection, and to selectively concatenate attitude solutions from the attitude history file or tape into a permanently mounted attitude transmission data set. Header records in the transmission data set are generated automatically by the interrogation and update utility. The contents of this data set are then transmitted to the IPD via the ADL or magnetic tape. After receipt of the attitude solutions by the IPD has been verified, the operator uses the interrogation and update utility to alter the corresponding summaries in the log data set to reflect the fact that the data have been successfully transmitted.

3.3 COMPUTER ENVIRONMENT

3.3.1 Hardware Requirements

3.3.1.1 Acquisition Mode (Near-Real-Time) Requirements

3.3.1.1.1 Acquisition Mode Nominal Support Requirements

The following equipment is required on the primary support computer during nominal acquisition mode:

Equipment	Quantity	Description
Graphic display unit (e.g., 2250)	1	Used for interactive processing
Graphic display unit (e.g., 2260)	1	Used for job scheduling and data set editing
Sharable disk pack (e.g., 2314)	1	Contains all data sets required for HCMM attitude determination processing
Line printer	1	Used for hardcopy output
350K bytes of computer core		

3.3.1.1.2 Acquisition Mode Backup Support Requirements

When the primary support computer is unavailable, acquisition mode attitude determination efforts for HCMM are performed on the backup computer. The same equipment required for nominal support is required for backup support.

3.3.1.2 Definitive Mode Requirements

3.3.1.2.1 Nominal Definitive Mode Support Requirements

The following equipment is required on the primary support computer during the nominal definitive mode:

Equipment	Quantity	Description
Graphics display unit (e.g., 2250)	1	Used for interactive processing

Equipment	Quantity	Description
Graphic display unit (e.g., 2260)	1	Used for job scheduling and data set editing
Line printer	1	Used for hardcopy output
350K bytes of computer core		

3.3.1.2.2 Definitive Mode Backup Support Requirements

The same equipment required for nominal definitive support on the primary support computer is required for backup definitive support on the backup computer.

3.3.1.3 Utility Requirements

The residual dipole bias utility and the scanner bias utility will operate graphically under GESS. Each requires one graphics display unit for interactive processing and for viewing input/output displays. The log interrogation and update utility requires one terminal device to view the contents of the log data set and to alter summaries in the data set.

3.3.2 Job Scheduling

During nominal and backup support, one initiator will be required; jobs will be submitted via deck or terminal.

3.3.3 Program Size

The HCMM ADS will execute in a maximum of 350K bytes of main storage.

3.4 OPERATIONAL CONSIDERATIONS

Because telemetry data is restricted to times of station contact, the HCMM attitude support system is a pass-oriented system. The three primary requirements of the support system are attitude monitoring, bias determination, and definitive attitude processing, and the anticipated operational atmosphere at these times is described below.

3.4.1 Attitude Monitoring

During the acquisition and orbit adjust phases of the mission, the support system will provide attitude monitoring in near-real time. When sufficient telemetry data has been received by MSOCC from the tracking station and written to the telemetry data set, the HCMM ADS will process the data in interactive mode. This processing may begin before all the telemetry data has been written to the data set if time-critical monitoring is desired, but only that data present at the time when the telemetry processor reads the data set will be processed. The monitoring support consists of GESS tabular displays and plots, as well as hardcopy printout of significant attitude parameters.

3.4.2 Bias Determination

During the orbit adjust phase and immediately thereafter, the HCMM ADS will use MSOCC telemetry data to generate a data base consisting of attitude control hardware data, attitude determination sensor data, and attitude solutions which will be used for determining the spacecraft residual dipole bias and the scanner pitch and roll constant biases.

The ADGEN/HCMM program requires high-quality attitude control hardware data, accurate attitude solutions, and accurate spacecraft ephemeris to estimate the spacecraft residual dipole bias effectively. Because attitude solutions and spacecraft ephemeris are likely to be less accurate during a 90-degree pitch maneuver, data will probably not be collected at these times. At other times, however, the HCMM ADS can write the electromagnet coil and scanwheel rate data to appropriate data sets and the attitude solutions to the attitude history file whenever the previously mentioned accuracy requirements are met for the pass being processed. The amount of data to be collected is TBD, but an initial estimate is approximately 10 passes of data. The ADGEN program will then analyze this data to obtain an estimate of the residual dipole bias. Further data can be collected if these estimates are not sufficiently accurate. Within 2 weeks of completion of the orbit adjust phase of the mission, the spacecraft residual dipole bias will be trimmed to less than 400 pole-centimeters.

The scanner bias utility requires high-quality Sun sensor and IR scanner data and accurate spacecraft ephemeris to effectively estimate the scanner pitch and roll constant biases. In addition, the data samples should be restricted to passes which are very similar to definitive mode passes and should provide a variety of Sun/spacecraft geometries. Therefore, data collection will be limited to those passes immediately following the orbit adjust phase which exhibit nominal attitude behavior. A station prediction program, which provides information about the Sun/spacecraft geometry at the time of expected station coverage, will be used to predict which passes contain geometrically useful sensor data. The amount of data to be collected is TBD, but an initial estimate is approximately 5 passes of data. The scanner bias utility will estimate the scanner pitch and roll constant biases within 2 weeks of completion of the orbit adjust phase of the mission.

3.4.3 Definitive Attitude Processing

Both the HCMM ADS and the log interrogation and update utility are used to meet the definitive attitude processing requirements. The ADS will process IPD telemetry data, write the definitive attitude solutions to the attitude history file, and write quality assurance and logging parameters to the log data set on a pass-by-pass basis. The ADS may be executed interactively or noninteractively for definitive attitude processing.

On a regular basis (perhaps daily), an analyst will use the log interrogation and update utility to verify the quality of the attitude solutions for each pass which has been processed since the last quality assurance check and to indicate which solutions may be transmitted to the IPD. An operator at GSFC will use the utility on a TBD basis (from approximately once per day to once per week) to move the validated attitude solutions to the transmission data set and concatenate these solutions.

SECTION 4 - ANALYTICAL CONSIDERATIONS

This section describes arithmetic and logical functions required for the HCMM ADS. Each of these functions is discussed to a level of detail which is required to support design efforts effectively. This requires the development of mathematical expressions for some functions and the presentation of decoding algorithms for others. Discussion of data set formats is deferred to Section 5.

For purposes of presentation, the functions of the Attitude Determination System are divided into four primary functions: telemetry processor, data adjustment, attitude determination, and output and logging. Section 4.1 discusses coordinate systems, Section 4.2 discusses telemetry processor functions, Section 4.3 discusses data adjustment functions, and Section 4.4 discusses attitude determination functions.

4.1 COORDINATE SYSTEMS

4.1.1 Geocentric Inertial Coordinate System

The geocentric inertial (GI) coordinate system is defined as true of date (TOD). The X-axis points to the vernal equinox; the Z-axis points to the celestial north pole; the Y-axis points in the direction of the vector cross product $\hat{Z} \times \hat{X}$. The X-Y plane of the system corresponds to the Earth's equatorial plane. The Earth's center is the center of the GI coordinate system.

4.1.2 Orbital Coordinate System

The orbital coordinate system, described in Figure 1-1 is a spacecraft-centered coordinate system. The Y (pitch)-axis is the negative orbit normal; the Z (yaw)-axis is the nadir or local vertical, positive toward the geocenter; the X (roll)-axis completes a right-handed system, i.e., $\hat{X} = \hat{Y} \times \hat{Z}$. For a circular orbit, the X-axis is the spacecraft velocity vector.

4.1.3 Spacecraft Coordinate System

The HCMM spacecraft (body) coordinate system is illustrated in Figure 1-3. The angular momentum of the scanwheel is along the spacecraft negative Y-axis.

4.1.4 Coordinate System Transformations

The transformation from the geocentric inertial to orbital coordinate system, denoted [R], is determined from the position $\bar{E}$ and velocity $\bar{V}$ of the spacecraft in GI coordinates. The orbit normal unit vector $\hat{n}$ is

$$\hat{n} = \frac{\hat{E} \times \hat{V}}{|\hat{E} \times \hat{V}|} \tag{4-1}$$

Then

$$[R] = \begin{bmatrix} -(\hat{E} \times \hat{n})^T \\ \hline -(\hat{n})^T \\ \hline -(\hat{E})^T \end{bmatrix} \tag{4-2}$$

The transformation from the orbital to the spacecraft coordinate system, denoted [K], is expressed in terms of an ordered 2-1-3 Euler transformation, where θ_2 is the pitch angle (denoted p), θ_1 is the roll angle (denoted r), and θ_3 is the yaw angle (denoted y).

$$[K] = \begin{bmatrix} k_{11} & k_{12} & k_{13} \\ k_{21} & k_{22} & k_{23} \\ k_{31} & k_{32} & k_{33} \end{bmatrix} \tag{4-3}$$

where $k_{11} = \cos p \cos y + \sin p \sin r \sin y$

$k_{12} = \cos r \sin y$

$k_{13} = -\sin p \cos y + \cos p \sin r \sin y$

$k_{21} = -\cos p \sin y + \sin p \sin r \cos y$

$k_{22} = \cos r \cos y$

$k_{23} = \sin p \sin y + \cos p \sin r \cos y$

$k_{31} = \sin p \cos r$

$k_{32} = -\sin r$

$k_{33} = \cos p \cos r$

If the elements of [K] are known, pitch, roll, and yaw are given by

$$r = \sin^{-1} (-k_{32}) \tag{4-4a}$$

$$p = \text{ATAN2} (k_{31}, k_{33}) \tag{4-4b}$$

$$y = \text{ATAN2} (k_{12}, k_{22}) \tag{4-4c}$$

Equation (4-4a) defines $-\pi/2 < r \leq \pi/2$, which means $\cos r \geq 0$. This fact, coupled with the FORTRAN function ATAN2, resolves quadrant ambiguities and defines $-\pi < p, y \leq \pi$.

For some applications, it is more convenient to express the orbital to spacecraft transformation in terms of rotations p', r', and y' about the orbital pitch, roll, and yaw axis, respectively. This is not equivalent to the 2-1-3 Euler transformation because the rotations are made about the original, not intermediate, coordinate axes. This transformation can be determined from successive applications of the following well-known result from vector algebra:

Given two vectors $\bar{a}$ and $\bar{b}$, the vector $\bar{c}$ which results by rotating $\bar{b}$ positively about $\bar{a}$ through an angle θ is

$$\bar{c} = (\bar{b} \cdot \bar{a})\,\bar{a} + \cos\theta\,(\bar{b} - (\bar{b} \cdot \bar{a})\,\bar{a}) + \sin\theta\,(\bar{a} \times \bar{b}) \qquad (4\text{-}5)$$

The resultant matrix is

This is simply 3-1-2 sequence!

$$
\begin{aligned}
k_{11} &= \cos y' \cos p' - \sin y' \sin p' \sin r' \\
k_{12} &= \cos y' \sin p' \sin r' + \sin y' \cos p' \\
k_{13} &= -\sin p' \cos r' \\
k_{21} &= -\sin y' \cos r' \\
k_{22} &= \cos y' \cos r' \qquad (4\text{-}6) \\
k_{23} &= \sin r' \\
k_{31} &= \cos y' \sin p' + \sin y' \cos p' \sin r' \\
k_{32} &= \sin y' \sin p' - \cos y' \cos p' \sin r' \\
k_{33} &= \cos r' \cos p'
\end{aligned}
$$

Of course all [illegible] are similar!

It should be noted that when pitch, roll, and yaw are near zero and the small angle approximation $\sin\theta \approx \theta$ and $\cos\theta \approx 1$ used, the two formulations for [K] (Equations (4-3) and (4-6)) are identical to first order. It should also be noted that the roll angle r' and the nadir angle η (the angle between the spacecraft Y-axis and the spacecraft to Earth vector) are related by

$$r' = \pi/2 - \eta \tag{4-7}$$

The transformation from GI to spacecraft coordinates, denoted [A], is the attitude matrix and is given by

$$[A] = [K]\,[R] \tag{4-8}$$

4.2 TELEMETRY PROCESSOR

4.2.1 Data Conversion

Appendix C contains a bit-by-bit description of the entire HCMM telemetry stream. This description includes information on how to interpret the PCM mode word and all status bits.

4.2.1.1 Magnetometer, Scanner, Magnetic Rate, and Electromagnetic Coil Data

For attitude determination purposes, the conversion of telemetry data to engineering units is a two-step process: first, the conversion from digital to analog; and then the conversion from analog to engineering units. The procedure for these conversions is as follows.

Telemetry words 53, 54, and 56 (the A/D CONV CAL Levels 1, 2, and 3, respectively) are unpacked, and let T3, T4, and T5 represent their values in decimal counts. Define the intermediate variables r, s, and t recursively by

$$r = \frac{V5 - V3}{(T5 - T3)\,(T5 - T4)} - \frac{V4 - V3}{(T4 - T3)\,(T5 - T4)}$$

$$s = \frac{V4 - V3}{T4 - T3} - r\ (T4 + T3)$$

$$t = V3 - r\ (T3)^2 - s(T3)$$

where V3, V4, and V5 are three reference voltages (approximately 0.11, 2.51, and 5.01) which are supplied via NAMELIST. Then the conversion from decimal counts, D, to a preliminary analog voltage, V_P, is given by

$$V_P = rD^2 + sD + t$$

The data types to be converted in this fashion are

- Magnetometer (pitch, roll, and yaw)
- Electromagnet bias (pitch, roll, and yaw)
- Electromagnet moment (pitch, roll, and yaw)
- Scanwheel speed (coarse and fine)
- Roll error (fine and coarse)
- Pitch error (fine and coarse)
- Duty cycle (I and II)
- Magnetic field rate (pitch, roll, and yaw)
- Roll bias
- 5- and 10-volt power supply voltages

4.2.1.1.1 Analog Data Correction

Onboard the spacecraft, certain data types use the 5- and 10-volt power supplies to generate the telemetered voltage. In the attitude determination process, the effects of fluctuations in the 5- and 10-volt power supplies must be removed from the preliminary analog voltage, V_P, to generate a net analog voltage, V_N, before further processing can be done.

- <u>5- and 10-volt power supply conversion.</u> The 5- and 10-volt supply readings, V_5 and V_{10}, are given by

$$V_5 = a_5 V_{P5} + b_5$$

$$V_{10} = a_{10} V_{P10} + b_{10}$$

where V_{P5} and V_{P10} are the preliminary analog voltages and a_5, b_5, a_{10}, and b_{10} are NAMELIST-controlled parameters.

- <u>Compensation for 5-volt power supply fluctuations.</u> For those data types which depend on the 5-volt power supply reading, the net voltage is

$$V_N = V_P - c_5(V_5 - 5)$$

where c_5 is a data-type-dependent NAMELIST parameter. The data types converted in this fashion are

  - Fine roll error
  - Magnetic field rate (pitch)
  - Magnetic field rate (roll)
  - Magnetic field rate (yaw)

- <u>Compensation for 10-volt power supply fluctuations.</u> For those data types which depend on the 10-volt power supply reading, the net voltage is

$$V_N = V_P - c_{10}(V_{10} - 10)$$

where c_{10} is a data-type-dependent NAMELIST parameter. The data types converted in this fashion are

- Magnetometer (pitch)
- Magnetometer (roll)
- Magnetometer (yaw)
- Electromagnet moment (pitch)
- Electromagnet moment (roll)
- Electromagnet moment (yaw)

For the following data types, the net voltage is the same as the preliminary voltage:

- Electromagnet bias (pitch, roll, and yaw)
- Coarse roll error
- Pitch error (fine and coarse)
- Fine scanwheel speed
- Coarse scanwheel speed
- Roll bias
- Duty cycle (I and II)

4.2.1.1.2 Net Analog to Engineering Unit Conversion

The net voltage for all data types, except for the electromagnet bias, is converted to engineering units, EU , by

$$EU = a\ V_N + b$$

where a and b are data-type-dependent NAMELIST parameters. (For purposes of this conversion, each axis of the magnetometer, electromagnet moment, and magnetic field rate is considered a separate data type.)

The electromagnetic bias voltage is converted to pole-centimeters by means of calibration points. For each axis, a set of NAMELIST-controlled calibration points (V_1, P_1), (V_2, P_1), ..., (V_7, P_7) is associated. The data conversion algorithm is

- If $V_N \leq V_1$, $EU = P_1$
- If $V_N \geq V_7$, $EU = P_7$
- If $V_k < V_N \leq V_{k+1}$, then $EU = \dfrac{(P_{k+1} - P_k)(V_N - V_k)}{V_{k+1} - V_k} + P_k$

4.2.1.1.3 Magnetometer Data Selection

If the Earth's magnetic field is approximated by a dipole along its spin axis, and if the spacecraft is assumed to be in a circular polar orbit, then the Earth's magnetic field $\bar{M}_O$ expressed in orbital coordinates is

$$\bar{M}_O = B \begin{pmatrix} \cos \omega_o t \\ 0 \\ 2 \sin \omega_o t \end{pmatrix}$$

where $\omega_o = 0.00108$ radians per second is the orbital angular velocity and $B = 230$ milligauss at 600 kilometers. In body coordinates, the magnetic field $\bar{M}_B$ is

$$\bar{M}_B = [K] \, \bar{M}_O$$

where $[K]$ is the transformation from orbital to body coordinates. The field rate in body coordinates, $\dot{\bar{M}}_B$, is

$$\dot{\bar{M}}_B = [\dot{K}] \, \bar{M}_O + [K] \, \dot{\bar{M}}_O = - \omega_B \times \bar{M}_B + [K] \, \dot{\bar{M}}_O$$

At the null attitude (pitch, roll, and yaw equal to zero), this equation simplifies to

$$\dot{\bar{M}}_B = B\begin{pmatrix}(\omega_o - 2\dot{p}) \sin \omega_o t \\ 2\dot{r} \sin \omega_o t - \dot{y} \cos \omega_o t \\ (\dot{p} + \omega_o) \cos \omega_o t\end{pmatrix}$$

where $\dot{p}$, $\dot{r}$, and $\dot{y}$ are the pitch, roll, and yaw rates, respectively. Clearly, the maximum field rate along the body X axis is $B(\omega_o - 2\dot{p})$ milligauss per second, the maximum rate along the body Z axis is $B(\dot{p} + \omega_o)$ milligauss per second, and the maximum rate along the body Y axis is $B(4\dot{r}^2 + \dot{y}^2)^{1/2}$ milligauss per second.

In mission mode, when the body rates are limited to 0.01 degree per second, the maximum change in the magnetic field during 1 second (the minor frame rate), $\Delta\bar{M}$, expressed in milligauss is

$$\Delta\bar{M} = \begin{pmatrix}.33 \\ .09 \\ .29\end{pmatrix}$$

The bucket size for the magnetic field readings is 4.68 milligauss, so that in mission mode for steady-state behavior, the magnetometer data will change buckets only infrequently (every 10 to 20 seconds), and therefore one need select only one of the four magnetometer readings available in each minor frame for further data processing.

In the early stages of acquisition mode, before rate capture has been achieved, body rates of several revolutions per minute are possible, and at these times the magnetic field may change by more than four buckets per second. However, it should be stressed that at this time the attitude determination system will be calculating solutions for satellite monitoring, not for definitive attitude, when accuracy requirements are less stringent. In addition, during acquisition the magnetometer biases will not be known. For these reasons, it is sufficient to select the magnetometer reading at the sampling time nearest to that of the Sun sensor data, which is available only once per minor frame.

4.2.1.2 Sun Sensor Data

The Sun data in the telemetry stream consist of two 8-bit bytes containing N_a and N_b , and three bits denoting the ID of the Sun sensor. The interpretation of the Sun sensor ID bits is given in Table 4-1.

Table 4-1. Sun Sensor Identification Code

ID	MEANING
XXXXX001	SUN SENSOR 1 ILLUMINATED
XXXXX010	SUN SENSOR 2 ILLUMINATED
XXXXX011	SUN SENSOR 3 ILLUMINATED
XXXXX111	NO SUN SENSOR ILLUMINATED

4.2.2 Spacecraft Clock and Attached GMT Checks

The times associated with the data are determined either from the telemetered spacecraft clock count or the Greenwich mean time (GMT) attached to the data by the tracking station. Checks on these values can be performed on three levels.

The first level is limit checking of the GMT and/or the spacecraft clock. In this level of checking, acceptable data must satisfy the following inequality:

$$GMTMIN \leq GMT_i \leq GMTMAX$$

where GMTMIN and GMTMAX are lower and upper limits, respectively, set through NAMELIST, and GMT_i is the attached GMT of the ith minor frame of data. A similar check can be made on the spacecraft clock value.

The second level of time checking is correlation of the attached GMT with the spacecraft clock on a frame by frame basis. Acceptable data must satisfy the equation

$$GMT_i = GMT_0 \pm UNCERT_1 + (TCG_i - TCG_0)\ (CLKPRD \pm UNCERT_2)$$

where
- GMT_i = attached GMT of the ith minor frame in seconds
- GMT_0 = a reference GMT corresponding to TCG_0 in seconds
- $UNCERT_1$ = uncertainty in GMT in seconds
- TCG_i = spacecraft clock count for the ith minor frame
- TCG_0 = the spacecraft clock reference corresponding to GMT_0
- CLKPRD = the spacecraft clock period in seconds per count
- $UNCERT_2$ = the uncertainty in CLKPRD in seconds per count

$UNCERT_1$, CLKPRD, and $UNCERT_2$ are assigned TBD default values, but can be changed through NAMELIST.

The third level of time checking is a search for self-consistent sets of the attached GMT or the spacecraft clock. For sets of GMT data to be considered self-consistent, they must satisfy the equation

$$GMT_j = GMT_i + (j - i)\ (FRMPRD \pm UNCERT) \qquad j > i$$

where GMT_j = the attached GMT of the jth minor frame in seconds

GMT_i = the attached GMT of the ith minor frame in seconds

j = the minor frame numbef of the jth minor frame

i = the minor frame number of the ith minor frame

FRMPRD = the minor frame period in seconds per count

UNCERT = a GMT tolerance in seconds

FRMPRD and UNCERT are assigned TBD default values and can be changed via NAMELIST. An analogous capability will optionally check for self-consistency in the spacecraft clock data.

All data in the largest set of self-consistent values are unflagged; the remaining data are flagged. Note that data contained in the largest self-consistent set need not occur consecutively in telemetry.

4.3 DATA ADJUSTMENT

4.3.1 Ephemeris

Sun, spacecraft, and magnetic field ephemeris are required to perform scalar tests for data validation functions (Reference 1). In addition, Sun and magnetic field vectors in the orbital coordinate system are calculated at user-specified time intervals (10 seconds is nominal) for use by the attitude determination functions.

Sun, spacecraft, and magnetic field ephemeris are required in true of date (TOD) inertial coordinates to an accuracy sufficiently greater than the Sun, horizon scanner, and magnetometer data accuracy so that the sensor errors dominate the error budget. Assuming an order of magnitude greater accuracy for the ephemeris, the following requirements are obtained:

- Spacecraft position and velocity: 0.05 degree (6 kilometers in track)

- Sun position: 0.025 degree
- Magnetic field direction: 0.05 degree

The spacecraft ephemeris accuracy is obtained from a definitive orbit tape in the EPHEM format. An internal orbit generator is required only for system testing. The Sun position vector requires updating only every 30 minutes and the parallax due to the spacecraft orbit (less than 0.03 degree) may be ignored. Magnetic field ephemeris is unavailable to the specified accuracy and the best available field model should be used.

The validity of the spacecraft ephemeris data is checked by comparing the orbit inclination obtained from the EPHEM header record with the nominal value of the inclination. If this difference is greater than some specified tolerance, it is assumed that the spacecraft ephemeris data is invalid.

4.3.2 IR Scanner Data Conversion

IR scanner data consists of the following: fine pitch, p_F ; coarse pitch, p_C ; fine roll, r_F ; coarse roll, r_C ; the duty cycle from acquisition of signal (AOS) to index, H_I ; and the duty cycle from index to loss of signal (LOS), H_O . These data can be used to determine the spacecraft rotations, p' and r' , about the orbital pitch and roll axes, respectively (see Equation (4-6)). The IR scanner data conversion process consists of

- Converting the duty cycle data to corresponding pitch and roll data
- Computing the crossing latitudes of the Earth scans
- Correcting the scanner data for oblateness, horizon radiance, and orbit eccentricity effects
- Finally, selecting one value for pitch and roll which reflects these corrections

4.3.2.1 Duty Cycle Conversion

The fundamental equations relating the nadir angle (angle between the spacecraft pitch axis and the spacecraft to Earth vector), IR scanner duty cycle, spacecraft altitude, CO_2 layer height, and triggering radius of the Earth are obtained from the spherical triangle shown in Figure 4-1:

$$\sin \rho_{I,O} = \frac{R_{EI,O} + h_{I,O}(\lambda)}{R} \tag{4-9a}$$

$$\cos \rho_{I,O} = \cos \eta \cos \gamma + \sin \eta \sin \gamma \cos \Omega_{I,O} \tag{4-9b}$$

where $\rho_{I,O}$ = angular radius of the Earth at AOS, LOS, respectively

R = distance from spacecraft to Earth's geocenter

$R_{EI}(\lambda)$ = radius of Earth at in-crossing latitude

$R_{EO}(\lambda)$ = radius of Earth at out-crossing latitude

$h_I(\lambda)$ = CO_2 layer height at in-crossing latitude

$h_O(\lambda)$ = CO_2 layer height at out-crossing latitude

η = nadir angle

γ = half angle of scan cone (= 45 degrees)

Ω_I = duty cycle from AOS to nadir

Ω_O = duty cycle from nadir to LOS

The roll angle and nadir angle are related by

$$r = \pi/2 - \eta \tag{4-10}$$

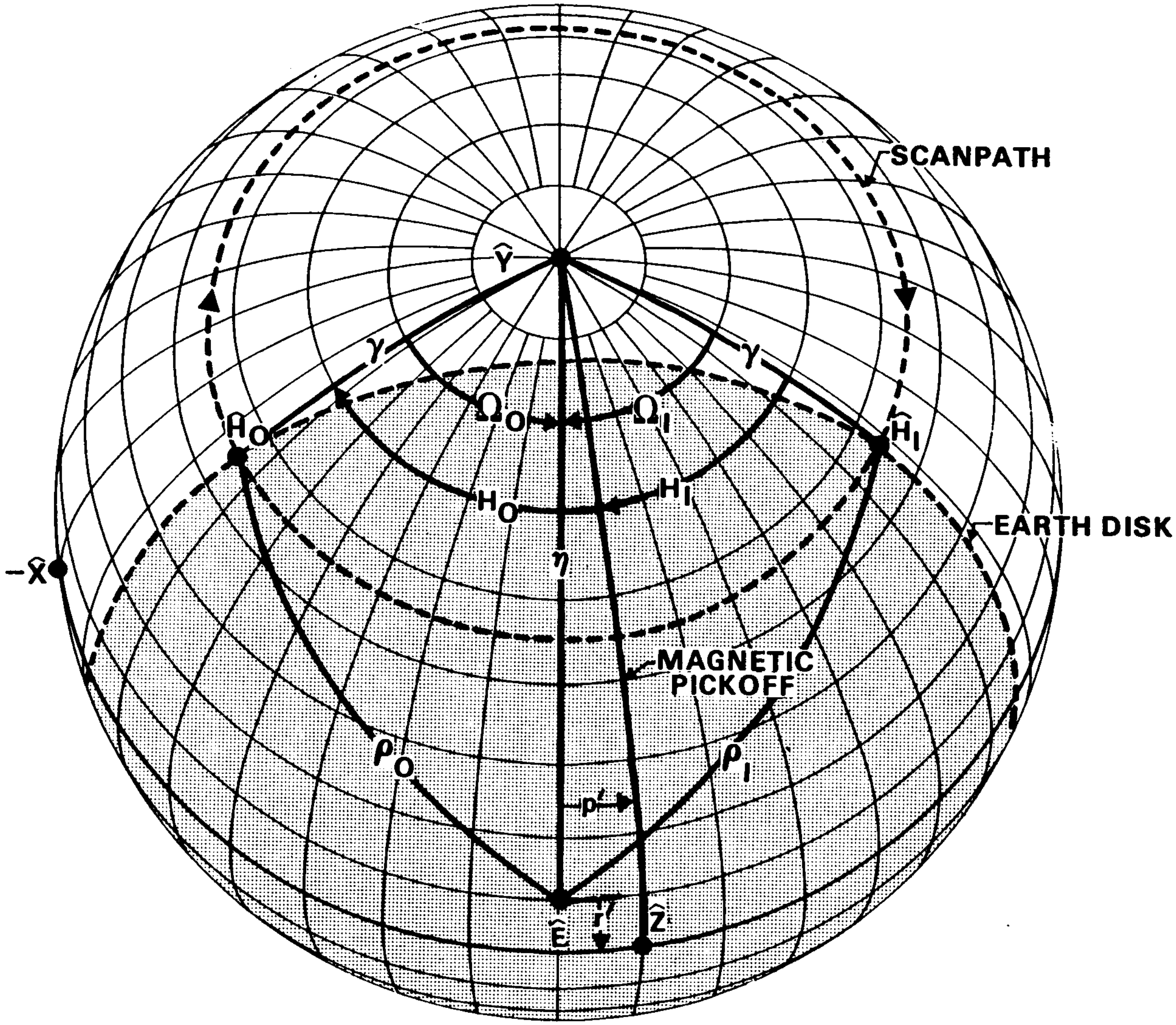


Figure 4-1. Scanner Pitch and Roll Angle Geometry

To convert the IR scanner duty cycle to an equivalent roll angle, r_D , Equations (4-9a) and (4-9b) are combined to yield

$$\frac{\left[R_N^2 - (R_{EN} + h_N)^2\right]^{1/2}}{R_N} = \cos\eta \cos\gamma + \sin\eta \sin\gamma \cos\left(\frac{H_I + H_O}{2}\right) \quad (4\text{-}11)$$

where R_N , R_{EN} , and h_N are nominal values for the distance from the spacecraft to the Earth's geocenter, the radius of the Earth, and the CO_2 layer height. For HCMM, the values used by Ithaco for calibration procedures are as follows:

$$\begin{aligned} R_N &= 6967.5 \text{ kilometers} \\ R_{EN} &= 6367.5 \text{ kilometers} \\ h_N &= 42.21 \text{ kilometers} \\ H_O = H_I &= 56.33 \text{ degrees} \end{aligned} \quad (4\text{-}12)$$

Equation (4-11) can be solved for the nadir angle η as follows. Let

$$\begin{aligned} d &= \left[\cos^2\gamma + \sin^2\gamma \cos^2\left(\frac{H_I + H_O}{2}\right)\right]^{1/2} \\ a &= \cos\gamma/d \\ b &= \sin\gamma \cos\left(\frac{H_I + H_O}{2}\right)\Big/d \\ c &= \frac{\left[R_N^2 - (R_{EN} + h_N)^2\right]^{1/2}}{R_N\, d} \end{aligned} \quad (4\text{-}13)$$

Note that a and b satisfy $a^2 + b^2 = 1$, so that

$$a = \cos w$$
$$b = \sin w \tag{4-14}$$

where $w = \text{ATAN2}(b, a)$

Equation (4-11) can now be written

$$\cos w \cos \eta + \sin w \sin \eta = c \tag{4-15a}$$

or

$$\cos (\eta - w) = c \tag{4-15b}$$

which has two solutions given by

$$\eta = w \pm \cos^{-1}(c) = \text{ATAN2}(b, a) \pm \cos^{-1}(c) \tag{4-16}$$

Because $a > 0$, it follows that $|\text{ATAN2}(b, a)| < \pi/2$. However, near the null attitude (pitch, roll, and yaw near zero), $\eta = \pi/2 - r \approx \pi/2$. Therefore, the solution obtained by using the minus sign in Equation (4-16) is spurious, and the correct solution is

$$\eta_D = \text{ATAN2}(b, a) + \cos^{-1}(c) \tag{4-17}$$

The corresponding analytic roll angle, r_A , is given by

$$r_A = \pi/2 - \eta_D \tag{4-18a}$$

while the duty cycle roll angle, r_D, is

$$r_D = r_A + \Delta r_D \tag{4-18b}$$

where $\Delta r_D \sim 2$ degrees is a user-specified correction for the dual-flake bolometer signal processing.

The pitch angle, p_D, corresponding to the duty cycle data is

$$p_D = (H_O - H_I)/2 \tag{4-19}$$

4.3.2.2 Computation of Crossing Latitudes of the Earth Scans

For the nominal spacecraft attitude in mission mode (pitch = roll = yaw = 0), the orbital to spacecraft transformation $[K]$ is the identity matrix, so that

$$[A] = [R] \tag{4-20}$$

where $[A]$ is the GI to spacecraft transformation and $[R]$ is the GI to orbital transformation, which is obtained from ephemeris information.

In spacecraft coordinates, the spin axis of the IR scanner, denoted $\hat{r}_B$, is

$$\hat{r}_B = (0, -1, 0)^T \tag{4-21}$$

so that

$$\hat{r}_I = [R]^T \begin{bmatrix} 0 \\ -1 \\ 0 \end{bmatrix} = \begin{bmatrix} -r_{21} \\ -r_{22} \\ -r_{23} \end{bmatrix} \tag{4-22}$$

The horizon crossing unit vectors $\hat{H}$, shown in Figure 4-2, satisfy

$$\begin{aligned} \hat{H} \cdot \hat{r}_I &= \cos (180 - \gamma) \\ \hat{H} \cdot \hat{E} &= \cos \rho_{I,O} \\ \hat{H} \cdot \hat{H} &= 1 \end{aligned} \tag{4-23}$$

where γ = half angle of the scan cone (~45 degrees)

$\rho_{I,O}$ = angular radius of the Earth at AOS and LOS, respectively

$\bar{E}$ = spacecraft to Earth vector in GI coordinates (kilometers)

For a spherical Earth of radius R_E, and a constant height of the CO_2 layer, h,

$$\rho_I = \rho_O = \sin^{-1} [(R_E + h)/|\bar{E}|] \tag{4-24}$$

There are two solutions to Equations (4-23) and (4-24), denoted $\hat{H}_1$ and $\hat{H}_2$, one corresponding to AOS and one corresponding to LOS. To determine which corresponds to AOS, let

$$\hat{r} = (r_{11}, r_{12}, r_{13})^T \tag{4-24a}$$

i.e., the first row of the inertial to orbital transformation matrix. If

$$\hat{H}_1 \cdot \hat{r} > 0 \tag{4-24b}$$

then $\hat{H}_1$ corresponds to AOS and $\hat{H}_1 = \hat{H}_I$ and $\hat{H}_2 = \hat{H}_O$.

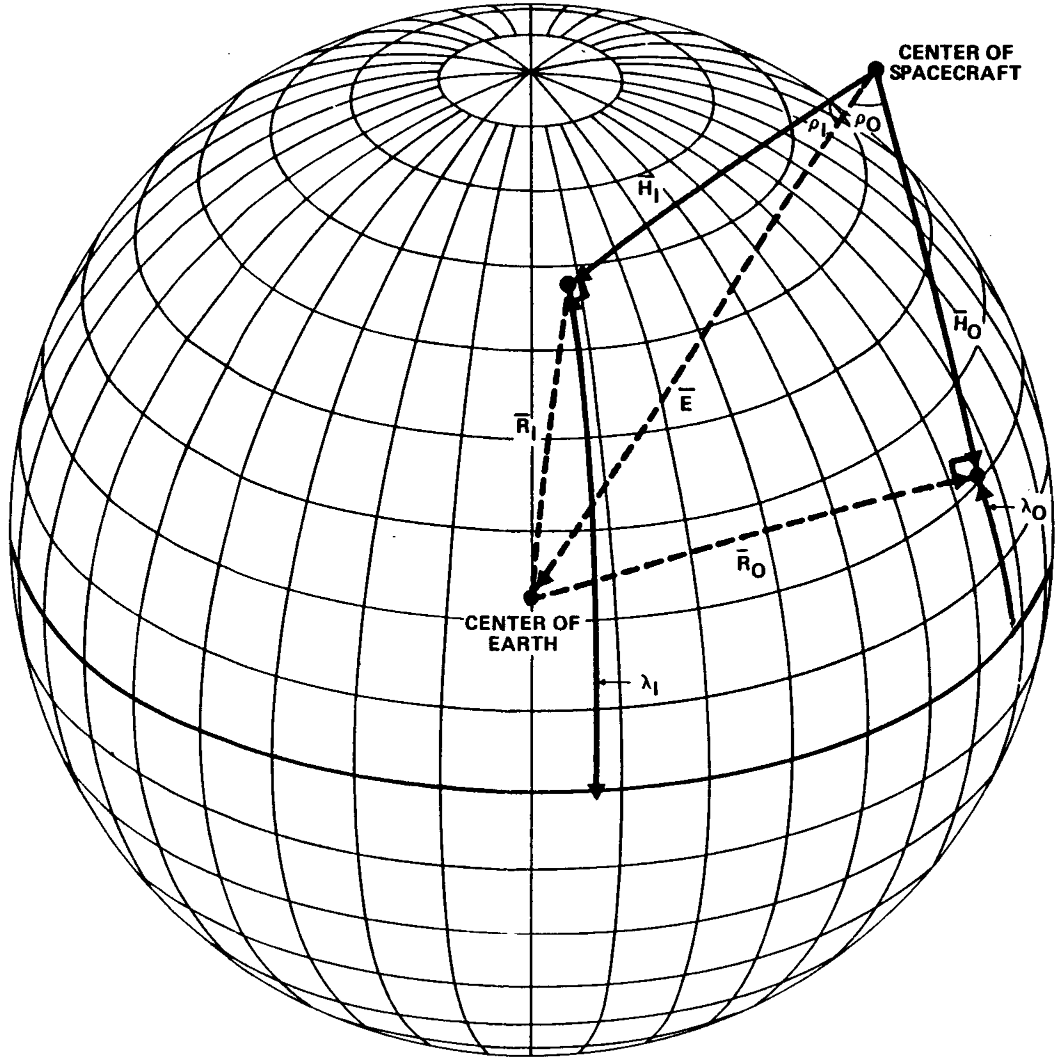


Figure 4-2. Horizon Crossing Geometry for a Spherical Earth

If

$$\hat{H}_1 \cdot \hat{r} < 0 \tag{4-24c}$$

then $\hat{H}_1$ corresponds to LOS and $\hat{H}_1 = \hat{H}_O$ and $\hat{H}_2 = \hat{H}_I$

If $\bar{R}_{I,O}$ denotes the vector from the center of the Earth to the horizon crossing points (Figure 4-2), then

$$\hat{R}_{I,O} = \hat{H}_{I,O}/\tan \rho - \hat{E}/\sin \rho \tag{4-25}$$

and the corresponding latitudes at AOS and LOS satisfy

$$\lambda_{I,O} = \sin^{-1}\left(\hat{R}_{I,O} \cdot (0, 0, 1)^T\right) \tag{4-26}$$

The effective triggering height of the infrared Earth at latitudes λ_I and λ_O is affected by Earth oblateness and the height of the CO_2 layer. Earth oblateness is modeled by

$$R_E(\lambda_{I,O}) = 6378.16\,(1 - C_1 \sin^2(\lambda_{I,O})) \tag{4-27}$$

where λ_I is the AOS latitude, λ_O is the LOS latitude, and $C_1 = 0.0033938$. The height of the CO_2 layer at AOS, $\lambda_I(\lambda_S, t)$, and at LOS, $\lambda_O(\lambda_S, t)$ depends both on the subsatellite point latitude, λ_S, direction of travel (north or south), and the season of the year. The CO_2 layer height is either specified as a constant or may be obtained by interpolating data from the horizon altitudes data set.

4.3.2.3 Oblateness, Eccentricity, and CO_2 Variation Corrections

Equation (4-9) implicitly defines Ω_O and Ω_I in terms of the remaining variables η , R , R_E , and h .

Implicitly differentiating with respect to η yields

$$0 = -\sin\eta \cos\gamma + \cos\eta \sin\gamma \cos(\Omega_{I,O}) - \sin\eta \sin\gamma \sin(\Omega_{I,O}) \frac{\partial\Omega_{I,O}}{\partial\eta} \tag{4-28}$$

Evaluating the partial derivative at the nominal values (Equation (4-12)) yields

$$\left.\frac{\partial\Omega_{I,O}}{\partial\eta}\right|_{NOM} = \frac{-\cos\gamma}{\sin\gamma \sin(\Omega_{I,O})} = \frac{-1}{\sin(\Omega_{I,O})} = -1.202 \tag{4-29}$$

Therefore,[1]

$$\Delta\Omega_{I,O} = -1.202\ \Delta\eta \tag{4-30}$$

Implicitly differentiating Equation (4-9) with respect to R yields

$$\frac{\frac{1}{2}\left[R^2 - (R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t))^2\right]^{-1/2} \cdot 2R}{R} - \frac{1}{R^2}\left[R^2 - (R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t))^2\right]^{1/2} \tag{4-31}$$

$$= -\sin\eta \sin\gamma \sin(\Omega_{I,O}) \frac{\partial\Omega_{I,O}}{\partial R}$$

---

[1] All evaluated partial derivatives are based on the nominal parameters used by Ithaco and are provided for analytical purposes only. Analytic expressions for all partial derivatives must be implemented.

Evaluating the partial derivative at the nominal values yields

$$\left.\frac{\partial \Omega_{I,O}}{\partial R}\right|_{NOM} = \frac{\frac{1}{R^2}\left[R^2 - \left(R_{EI,O}(\lambda) + h_{I,O}(\lambda)\right)^2\right]^{1/2} - \left[R^2 - \left(R_{EI,O}(\lambda) + h_{I,O}(\lambda)\right)^2\right]^{-1/2}}{\sin\eta \sin\gamma \sin\left(\Omega_{I,O}\right)} \tag{4-32}$$

$$= -0.000526$$

Therefore,

$$\Delta\Omega_{I,O} = -0.000526 \left(\frac{\text{radians}}{\text{kilometer}}\right) \Delta R = -0.03014 \left(\frac{\text{degree}}{\text{kilometer}}\right) \Delta R \tag{4-33}$$

Implicitly differentiating Equation (4-9) with respect to R_E yields

$$\frac{\frac{1}{2}\left[R^2 - \left(R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t)\right)^2\right]^{-1/2} (-2) \left(R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t)\right)}{R} = -\sin\eta \sin\gamma \sin\left(\Omega_{I,O}\right) \frac{\partial \Omega_{I,O}}{\partial R_E} \tag{4-34}$$

Evaluating the partial derivative at the nominal values yields

$$\left.\frac{\partial \Omega_{I,O}}{\partial R_{EI,O}}\right|_{NOM} = \frac{\left(R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t)\right)\left[R^2 - \left(R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t)\right)^2\right]^{-1/2}}{R \sin\eta \sin\gamma \sin\left(\Omega_{I,O}\right)} = 0.000572 \tag{4-35}$$

Therefore,

$$\left.\frac{\partial \Omega_{I,O}}{\partial R}\right|_{NOM} = \frac{\frac{1}{R^2}\left[R^2 - \left(R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t)\right)^2\right]^{1/2} - \left[R^2 - \left(R_{EI,O}(\lambda) + h_{I,O}(\lambda_S, t)\right)^2\right]^{-1/2}}{\sin\eta \sin\gamma \sin\left(\Omega_{I,O}\right)} \tag{4-36}$$

$$= -0.000526$$

Similarly,

$$\Delta\Omega_{I,O} = 0.000572 \left(\frac{\text{radian}}{\text{kilometer}}\right) \Delta h(\lambda) = 0.03277 \left(\frac{\text{degree}}{\text{kilometer}}\right) \Delta h(\lambda) \quad (4\text{-}37)$$

Since $r = \pi/2 - \eta$, it follows that

$$\Delta\Omega_{I,O} = 1.202\ \Delta r \quad (4\text{-}38)$$

Substituting this into Equations (4-33), (4-36), and (4-37) yields

$$\begin{aligned} \frac{\Delta r}{\Delta R} &= -0.02507 \left(\frac{\text{degree}}{\text{kilometer}}\right) \\ \frac{\Delta r}{\Delta R_E} &= 0.02726 \left(\frac{\text{degree}}{\text{kilometer}}\right) \\ \frac{\Delta r}{\Delta h} &= 0.02726 \left(\frac{\text{degree}}{\text{kilometer}}\right) \end{aligned} \quad (4\text{-}39)$$

From Equation (4-19), it follows that

$$\Delta p = \frac{1}{2} (\Delta\Omega_O - \Delta\Omega_I) \quad (4\text{-}40)$$

The roll data (r_F, r_C, and r_D) are corrected for oblateness, eccentricity, and horizon radiance variation effects to obtain corrected data, denoted by the overbar, by

$$\begin{aligned} \bar{r} = r - \frac{\Delta r}{\Delta R}(R - R_N) - \frac{1}{2} \Big\{ &\frac{\Delta r}{\Delta R_E} (R_{EI}(\lambda) + R_{EO}(\lambda) - 2\ R_{EN}) \\ &+ \frac{\Delta r}{\Delta h} (h_I(\lambda_S, t) + h_O(\lambda_S, t) - 2\ h_N) \Big\} \end{aligned} \quad (4\text{-}41)$$

where R_N = 6968 kilometers is the nominal semimajor axis of the orbit

R_{EN} = 6368 kilometers is the mean Earth radius

h_N = 42.21 kilometers is the nominal height of the CO_2 layer

Values for the difference quotients are given in Equation (4-39). Similarly, the pitch data (p_F, p_C, and p_D) are corrected for these same effects to obtain corrected data by

$$\bar{p} = p - \frac{1}{2}\,\frac{\Delta\Omega_{I,O}}{\Delta R_E(\lambda)}\left[R_{EO}(\lambda) - R_{EI}(\lambda) + h_O(\lambda_S, t) - h_I(\lambda_S, t)\right] - p_T \qquad (4\text{-}42)$$

where

$$\frac{\Delta\Omega_{I,O}}{\Delta R_E(\lambda)} = \left.\frac{\partial\,\Omega_{I,O}}{\partial\,R_{EI,O}}\right|_{NOM} \approx 0.03278 \text{ degree/kilometer} \qquad (4\text{-}43)$$

(see Equation (4-35), and p_T is a NAMELIST variable which compensates for the transfer function in the scanner electronics.

4.3.2.4 Pitch and Roll Angle Selection

After the effects of oblateness, orbit eccentricity, and horizon radiance variations have been compensated for, one value of the pitch angle, p', and one value of the roll angle, r', must be selected from the three corrected pitch angles ($\bar{p}_F$, $\bar{p}_C$, $\bar{p}_D$) and roll angles ($\bar{r}_F$, $\bar{r}_C$, $\bar{r}_D$), respectively. These selections are not made on a point-by-point basis, but rather based on the quality of data for the entire pass. If the magnitude of the fine pitch data is less than the fine pitch saturated value, p_{FSAT}, for a specified minimum percentage of time, the fine pitch data is selected. If not, the coarse pitch data is examined, and if its magnitude is less than a specified maximum pitch angle, p_{MAX}, for a specified minimum percentage of time, the coarse pitch data is selected. Finally, if neither criterion is satisfied, the duty cycle data is compared to the maximum pitch angle and selected in the same way as the coarse pitch data. If all criteria fail, no pitch data is accepted. If $\bar{p}$ denotes the pitch data which is accepted, the final pitch angle, p', is given by

$$p' = \bar{p} - p_B + k_p\, 90 \tag{4-44a}$$

where p_B is the constant pitch bias obtained from the scanner bias determination utility for the type of pitch data (fine, coarse, or duty cycle) which was selected and $k_p = 0$ if there is no pitch maneuver, $= 1$ if there is a +90-degree pitch maneuver, and $= -1$ if there is a -90 degree maneuver. The roll data is selected in an analogous manner, using different values for the saturation level, r_{SAT}; maximum angle, r_{MAX}; minimum percentage values; and roll bias, r_B, to obtain the final roll angle

$$r' = \bar{r} - r_B + r_p \tag{4-44b}$$

where r_p is a constant ($\approx$ 2.90 degrees) to be added in the case of a ± 90-degree pitch maneuver.

4.3.3 Computation of Body Vectors

IR scanner pitch and roll data can be used to obtain the zenith or Earth to spacecraft unit vector. In orbital coordinates this vector is

$$\hat{E}_O \equiv (0,\ 0,\ -1)^T \tag{4-45}$$

The zenith vector is rotated into the body frame using the orbital to body coordinate transformation $[K]$ (Equation 4-6)) which depends on the spacecraft attitude,

$$\hat{E}_B = [K]\,\hat{E}_O = (\sin p' \cos r',\ -\sin r',\ -\cos r' \cos p')^T \tag{4-46}$$

The magnetometer data, $\overline{M}_S$, is converted to the magnetic field in body coordinates by the linear transformation

$$\overline{M}_B = [M]\,\overline{M}_S + \overline{b} \tag{4-47}$$

where $\overline{b}$ is the vector of magnetometer biases and $[M]$ is the alignment matrix given by

$$[M] = \left[\hat{M}_{XB} \,\vdots\, \hat{M}_{YB} \,\vdots\, \hat{M}_{ZB}\right] \tag{4-48}$$

where $\hat{M}_{XB}$, $\hat{M}_{YZ}$, and $\hat{M}_{ZB}$ are unit vectors describing the orientation of the X-, Y-, and Z-axis magnetometers, respectively, in the body (nominally $[M] = [I]$) .

The Sun vector in body coordinates is calculated from Sun sensor N_a , N_b , and ID in the following manner. First define the intermediate variables x , y , t_α , t_β , and R by

$$x \equiv k_{x1} N_a - k_{x2} \tag{4-49a}$$

$$y \equiv k_{y1} N_b - k_{y2} \tag{4-49b}$$

$$t_\alpha = [nx/R] \tag{4-49c}$$

$$t_\beta = [ny/R] \tag{4-49d}$$

$$R = \left[h^2 - (n^2 - 1)(x^2 + y^2)\right]^{1/2} \tag{4-49e}$$

where n is the index of refraction of the Sun sensor slab, h is the thickness of the slab, and k_{x1}, k_{x2}, k_{y1}, k_{y2} are conversion constants. Nominal values for these parameters are

$$\begin{aligned} k_{x1} &= k_{y1} = 0.00275 \\ k_{x2} &= k_{y2} = 0.350625 \\ h &= 0.448 \\ n &= 1.4553 \end{aligned} \tag{4-50}$$

The Sun unit vector in sensor coordinates (Figure 4-3), $\hat{S}_S$, is

$$\hat{S}_S = \left(1 + t_\alpha^2 + t_\beta^2\right)^{-1/2} \left(-t_\alpha, t_\beta, 1\right)^T \tag{4-51}$$

The transformation from sensor to body coordinates is described by a Euler 3-1-3 rotation. If the three spacecraft to sensor rotation angles are θ_1, θ_2,

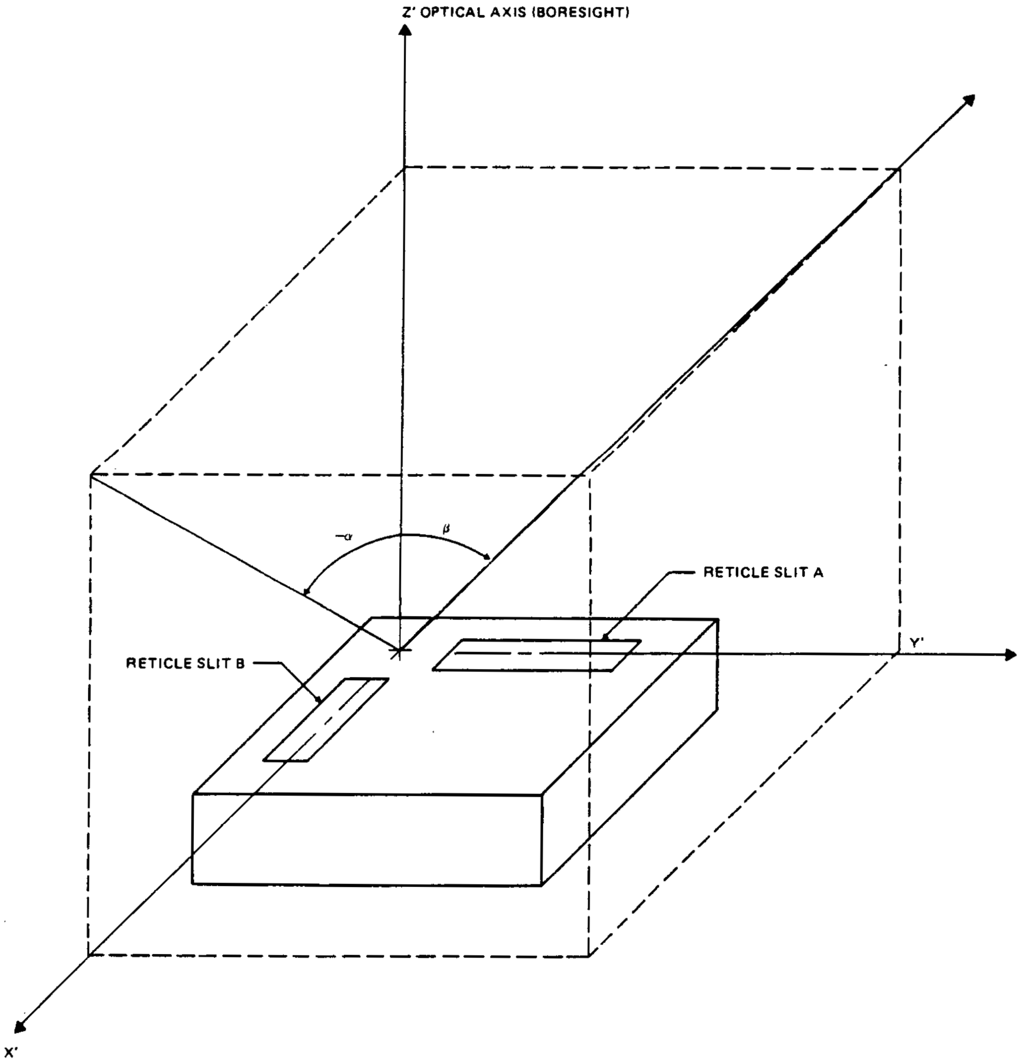


Figure 4-3. Sun Sensor Coordinate System

and θ_3, respectively, then transformation of the matrix for the ith (ID ≡ i) Sun sensor is

$$[S_i] = \begin{pmatrix} c\theta_3\, c\theta_1 - s\theta_3\, c\theta_2\, s\theta_1 & -c\theta_1\, s\theta_3 - c\theta_2\, c\theta_3\, s\theta_1 & s\theta_1\, s\theta_2 \\ s\theta_1\, c\theta_3 + s\theta_3\, c\theta_2\, c\theta_1 & -s\theta_3\, s\theta_1 + c\theta_2\, c\theta_3\, c\theta_1 & -s\theta_2\, c\theta_1 \\ s\theta_2\, s\theta_3 & c\theta_3\, s\theta_2 & c\theta_2 \end{pmatrix} \quad (4\text{-}52)$$

The nominal values of θ_1, θ_2, and θ_3 for each Sun sensor are given in Section 1.2.1.

The Sun vector in body coordinates computed from data from the ith Sun sensor is

$$\hat{S}_B = [S_i]\, \hat{S}_S \quad (4\text{-}53)$$

4.3.4 Data Validation

4.3.4.1 Attitude-Independent Checks

4.3.4.1.1 Limit Checks Involving One Data Type

The difference between the magnitude of the magnetic field in body coordinates, calculated from magnetometer data, and the magnitude of the inertial magnetic field is

$$\Delta M = \Big| \|\bar{M}_B\| - \|\bar{M}_I\| \Big| \quad (4\text{-}54)$$

Magnetometer data is flagged if ΔM exceeds a user-specified limit, i.e.,

$$\Delta M > \epsilon_M \quad (4\text{-}55)$$

4.3.4.1.2 Dot Product Check

The second type of validation check which requires no a priori estimate of the attitude is the dot product check. For this check, the angle between two reference vectors in inertial coordinates (obtained from ephemeris) is compared to the angle between these same reference vectors in body coordinates, which have been computed from appropriate sensor data. The difference in these angles for various data type combinations are

$$\Delta_1 = |\cos^{-1} (\hat{M}_B \cdot \hat{S}_B) - \cos^{-1} (\hat{M}_I \cdot \hat{S}_I)| \quad \text{(Sun/magnetometer)} \tag{4-56a}$$

$$\Delta_2 = |\cos^{-1} (\hat{E}_B \cdot \hat{S}_B) - \cos^{-1} (\hat{E}_I \cdot \hat{S}_I)| \quad \text{(IR/Sun)} \tag{4-56b}$$

$$\Delta_3 = |\cos^{-1} (\hat{E}_B \cdot \hat{M}_B) - \cos^{-1} (\hat{E}_I \cdot \hat{M}_I)| \quad \text{(IR/magnetometer)} \tag{4-56c}$$

These angular differences, which are evaluated whenever the appropriate sensor data are available, are compared to user-supplied tolerances ϵ_1, ϵ_2, and ϵ_3, respectively. The ith data type combination fails this test if

$$\Delta_i > \epsilon_i \tag{4-57}$$

If a data type combination fails this test, other validation checks are required to indicate which of the two possible sensor data types is the cause of this failure.

4.3.4.2 Attitude-Dependent Checks

Attitude-dependent checks are applied only during mission mode when the control system will maintain the pitch and roll angles within 1 degree of null and the yaw angle within 2 degrees of null. The basis of this check is to compare a reference vector in body coordinates, obtained from sensor data, to an estimate of this vector, obtained from the corresponding inertial vector and the a priori estimate of the null attitude.

If $\bar{A}_B$ is a reference vector in body coordinates and $\bar{A}_I$ is the corresponding inertial reference vector, then $\bar{A}_B$ and $\bar{A}_I$ are related by

$$\bar{A}_B = [K]\,[R]\,\bar{A}_I = [K]\,\bar{A}_O \tag{4-58}$$

where $\bar{A}_O$ is the reference vector expressed in orbital coordinates. For the null attitude,

$$[K] = [I] \tag{4-59}$$

so that the estimated value of $\bar{A}_B$ for the null attitude is $\bar{A}_O$.

The magnetometer data and the magnetometer biases provide a measure of the magnetic field in body coordinates, $\bar{M}_B$. Because attitude solutions can be obtained even if only one or two components of $\bar{M}_B$ are available, the null attitude check for magnetometer data is provided for each axis separately. The differences between orbital and body vector components are

$$\Delta_{M1} = |M_{B1} - M_{O1}| \tag{4-60a}$$

$$\Delta_{M2} = |M_{B2} - M_{O2}| \tag{4-60b}$$

$$\Delta_{M3} = |M_{B3} - M_{O3}| \tag{4-60c}$$

The ith component of the magnetometer data fails the null attitude check and is flagged if

$$\Delta_{Mi} > \epsilon_{Mi} \tag{4-61}$$

where ϵ_{Mi} is a user-specified tolerance. ϵ_{Mi} should be sufficiently large to allow for expected errors in the magnetometer data (analog-to-digital conversion and uncertainties in the magnetometer biases), small deviations from the null attitude, and errors in the computed inertial magnetic field.

For Sun sensor data, the following expression is evaluated:

$$\|\hat{S}_B - \hat{S}_O\|^2 = \|\hat{S}_B\|^2 + \|\hat{S}_O\|^2 - 2\hat{S}_B \cdot \hat{S}_O = 2(1 - \cos\theta_S) \approx \theta_S^2 \tag{4-62}$$

Here θ_S is the angle between the orbital and body Sun unit vector. The Sun data is flagged if

$$\theta_S > \epsilon_S \qquad \text{(4-63)}^1$$

where ϵ_S is user supplied.

4.3.5 Data Smoothing for Validation

The capability to smooth data is required both for data validation and for reducing the data volume. For validation purposes, this is a check to ensure that all data points in a given set are self-consistent. Smoothing for data reduction is useful if the data volume is too low for effective preaveraging.

[1] Equations (4-64) through (4-66) have been deleted.

Data types to be smoothed are fit to a set of Chebyshev polynomials (Reference 1) in the following way. Let T_i be the time of a valid data sample and y_i be its measured value at this time. The smoothed value, y_i, at this time for a polynomial of degree r is

$$\bar{y}_i = \sum_{k=0}^{r} c_k \, g_k(T_i) \tag{4-67}$$

where

$$T_i = \frac{2t_i - (t_o + t_n)}{t_n - t_o}$$

where g_k is the kth order Chebyshev polynomial, t_o and t_n are the minimum and maximum times respectively of the data points to be smoothed, and the constants c_0, c_1, ..., c_r are the coefficients determined by the fitting algorithm. The sum of the squared deviations is

$$S_r = \sum_{i=1}^{n} (y_i - \bar{y}_i)^2 \; . \tag{4-68}$$

and the standard deviation of the fit is

$$\sigma_r = \left(\frac{S_r}{n-1}\right)^{1/2} \tag{4-69}$$

In smoothing for validation, a data point y_i is rejected if $|y_i - \bar{y}_i| > m\,\sigma_r$, where m is the maximum number of standard deviations a valid data point will be allowed to deviate from the fit.

As the degree r of the Chebyshev polynomial increases, the more closely the polynomial approximates the data. With regard to validation, if the degree of the Chebyshev polynomial is too small, some valid data may be erroneously discarded. On the other hand, if the degree is too large, the polynomial may follow the noisy or invalid data. One method of selecting the degree of the polynomial is to choose r to be the smallest number satisfying

$$S_r < \epsilon S_0 \tag{4-70}$$

where ϵ is a constant, varying typically between 10^{-3} and 10^{-5}.

After the proper degree of the polynomial has been selected, invalid data are flagged and rejected. This process of computing a set of coefficients using valid data and then rejecting data which deviate too much from the fitting polynomial is continued until no more data are rejected. The final set of coefficients $c_0, c_1, c_2, \ldots, c_r$ is also used to generate sensor data if the option to smooth the data for output purposes is chosen.

4.3.6 Data Preaveraging

Attitude rates in on-orbit mode are expected to be about 0.01 degreee per second and the nominal attitude sensor data rate is once per second. Attitude solutions can be calculated less frequently, approximately once every 10 seconds, because the attitude nominally changes by at most 0.1 degree during this time. Therefore, the data volume may be reduced. This data reduction, which is optional, will be done either by preaveraging or by smoothing. The smoothing algorithm has already been discussed in the validation section. The preaveraged value of a data type is simply its average value in a time interval whose length is equal to the attitude solution period. The processing interval is divided into N such time intervals (bins), as described in Figure 4-4, centered at times $t_1, t_2, \ldots, t_N$.

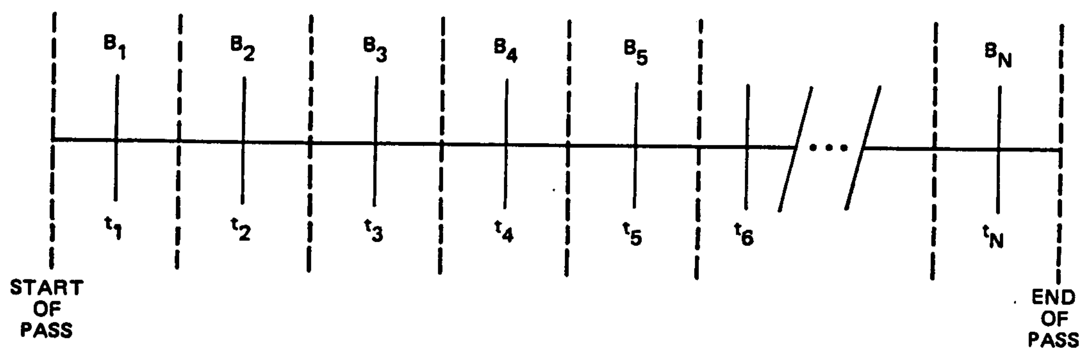


Figure 4-4. Division of Pass Into Bins for Preaveraging

If there are p valid data points within a bin B_j , then the preaveraged value $\bar{X}_j$ of a data type within that bin is

$$\bar{X}_j = \frac{1}{p} \sum_{i=1}^{p} X(t_i) \tag{4-71}$$

where all the times are contained in the bin B_j centered at the time t_j .

4.3.7 Calculation of Quality Assurance Parameters

In definitive mode, various quality assurance parameters are computed at the end of the pass. For Sun, scanner, and magnetometer data, the percent of valid data remaining and the largest gap in each data type after the completion of all data adjustment functions are computed.

Various quantitative measures of the valid data are also computed. If the deviation of a certain variable from its nominal value at time t_i is $X(t_i)$, then the rms deviation is

$$\mathrm{rms}(X) = \left(\frac{\sum_{i=1}^{N} X^2(t_i)}{N - 1} \right)^{1/2} \tag{4-72}$$

where N is the number of valid data points in the pass. This rms deviation is computed for the following variables:

- ΔM (Equation (4-54) Magnitude of magnetic field check
- Δ_1 (Equation (4-56a)) Sun/magnetometer dot product check
- Δ_2 (Equation (4-56b)) IR/Sun dot product check
- Δ_3 (Equation (4-56c)) IR/magnetometer dot product check
- Δ_{M1} (Equation (4-60a)) Magnetometer X-axis null attitude check
- Δ_{M2} (Equation (4-60b)) Magnetometer Y-axis null attitude check

Δ_{M3} (Equation (4-60c)) Magnetometer Z-axis null attitude check

θ_S (Equation (4-62)) Sun sensor null attitude check

The angular deviation from null as measured by IR scanner data can be defined in a manner analogous to that for Sun sensor data, Equation (4-62), using the Earth vectors in orbital and body coordinates.

This angle, θ_{IR}, is given by

$$\theta_{IR}^2 = \|\hat{E}_B - \hat{E}_O\|^2 \tag{4-73}$$

and is computed for quality assurance purposes.

4.3.8 Sun Sensor Field of View Display

The following algorithm shall be employed in developing the Sun sensor field of view (FOV) display.

The sunline in the FOV of a Sun sensor may be identified by the angle α between the sensor optical axis and the projection of the sunline onto the sensor X - Z plane and by the angle β between the sensor optical axis and the projection of the sunline on the sensor Y - Z plane (see Figure 4-3).

If $\hat{S}_s = (X_s, Y_s, Z_s)^T$ is the unit vector in the direction of the sunline, expressed in sensor coordinates, and α and β the corresponding angles, then $\hat{S}_s$ and α and β are related by

$$\cos \beta = \frac{Z_s}{\left(Y_s^2 + Z_s^2\right)^{1/2}}$$

$$\cos \alpha = \frac{Z_s}{\left(X_s^2 + Z_s^2\right)^{1/2}}$$

where $X_s = \sin\alpha \cos\beta \, (\sin^2\alpha \cos^2\beta + \cos^2\alpha)^{1/2}$

$Y_s = \cos\alpha \sin\beta \, (\sin^2\alpha \cos^2\beta + \cos^2\alpha)^{1/2}$

$Z_s = \cos\alpha \cos\beta \, (\sin^2\alpha \cos^2\beta + \cos^2\alpha)^{1/2}$

The four edges of the FOV for a given Sun sensor are defined by the following (all units are in degrees):

$$\alpha = 64, \quad \beta \in [-64, 64]$$

$$\alpha = -64, \quad \beta \in [-64, 64]$$

$$\beta = 64, \quad \alpha \in [-64, 64]$$

$$\beta = -64, \quad \alpha \in [-64, 64]$$

If $\hat{S}_s$ is a unit vector in the FOV for the ith Sun Sensor (i = 1, 2, 3), then the corresponding unit vector in spacecraft coordinates, denoted $\hat{S}_B = (X_B, Y_B, Z_B)^T$, is

$$\hat{S}_B = \left[S_i\right] \hat{S}_s$$

where $\left[S_i\right]$ is the sensor-to-body transformation matrix (Equation (4-52)). For display purposes, it is convenient to consider the coordinate system resulting by rotating the spacecraft coordinates system 90 degrees about the spacecraft Y axis. If $\hat{S}_B$ is a unit vector in spacecraft coordinates, then the corresponding unit vector in the rotated coordinate system, denoted $\hat{S}_R = (X_R, Y_R, Z_R)^T$, is

$$(X_R, Y_R, Z_R)^T = (-Z_B, Y_B, X_B)^T \qquad \text{(4-73a)}$$

The field of view display should plot the right ascension and declination, in the rotated coordinate system, for the following quantities:

- Edge of the FOV for all three Sun sensors
- Center of the FOV for all three Sun sensors
- Sun unit vector trajectory at the output frequency (orbit dependent)

The field of view display should also indicate Earth blockage by plotting the edge of the Earth disk. This is done in the following way. Spacecraft ephemeris defines the length of the Earth to spacecraft vector, denoted $|\bar{E}|$. The nominal angular radius of the Earth ρ is given by

$$\rho = \sin^{-1}\left[R_{EN} / |\bar{E}| \right]$$

where R_{EN} is the nominal radius of the Earth. In spacecraft coordinates, the edge of the Earth is described by a cone of unit vectors, $\hat{e}\ (\theta) = (e_x, e_y, e_z)^T$, defined by the azimuth angle θ

$$e_x = \cos\theta \sin\rho$$

$$e_y = \sin\theta \sin\rho$$

$$e_z = \cos\rho$$

$$0 \leq \theta \leq 2\pi$$

These vectors may then be transformed to the rotated coordinate system as in Equation (4-73a).

4.4 ATTITUDE DETERMINATION

4.4.1 Discrete Attitude Determination

4.4.1.1 Nominal Attitude Determination

The method of discrete attitude determination employed by the HCMM attitude determination system requires two reference unit vectors, each expressed both in GI and spacecraft coordinates, and the GI to orbital transformation. This information permits computation of the transformation from orbital to spacecraft coordinates, from which the Euler 2-1-3 (pitch, roll, and yaw) angles can be computed by Equation (4-4). This technique can be used for any combination of two sensor data types, each of which measures a different reference unit vector in spacecraft coordinates.

Let $\hat{a}_B$ and $\hat{b}_B$ be the two reference unit vectors in spacecraft coordinates and $\hat{a}_I$ and $\hat{b}_I$ the corresponding vectors in GI coordinates. Let [R] be the GI to orbital transformation computed from ephemeris data. Let $\hat{a}$ be the primary reference vector, which is the vector obtained from the more accurate sensor. Define the unit vectors $\hat{c}_B$, $\hat{d}_B$, $\hat{c}_I$, and $\hat{d}_I$ by

$$\begin{aligned} \hat{c}_B &= \hat{a}_B \times \hat{b}_B / |\hat{a}_B \times \hat{b}_B| \\ \hat{c}_I &= \hat{a}_I \times \hat{b}_I / |\hat{a}_I \times \hat{b}_I| \end{aligned} \tag{4-74}$$

and

$$\begin{aligned} \hat{d}_B &= \hat{a}_B \times \hat{c}_B \\ \hat{d}_I &= \hat{a}_I \times \hat{c}_I \end{aligned} \tag{4-75}$$

$\hat{c}$ and $\hat{d}$ are well-defined unless $\hat{a}$ and $\hat{b}$ are colinear, in which case three-axis attitude determination is impossible because $\hat{a}$ and $\hat{b}$ provide redundant information. Define the corresponding vectors in orbital coordinates by

$$\hat{a}_O = [R]\,\hat{a}_I$$
$$\hat{c}_O = [R]\,\hat{c}_I \qquad (4\text{-}76)$$
$$\hat{d}_O = [R]\,\hat{d}_I$$

and note that $\hat{a}$, $\hat{c}$, and $\hat{d}$ are mutually orthogonal in both coordinate systems. The orbital to spacecraft transformation K satisfies

$$[K]\,[O] = [B] \qquad (4\text{-}77)$$

where $[O] = [\hat{a}_O \,\vdots\, \hat{c}_O \,\vdots\, \hat{d}_O]$ and $[B] = [\hat{a}_B \,\vdots\, \hat{c}_B \,\vdots\, \hat{d}_B]$.

Because $[O]$ and $[B]$ are orthogonal transformations, i.e., their inverses and transposes are identical, $[K]$ can be determined by

$$[K] = [B]\,[O]^T \qquad (4\text{-}78)$$

The 2-1-3 Euler (pitch, roll, and yaw) angles are then given by Equation (4-4).

The reference vectors which may be used for discrete attitude determination are the Earth to spacecraft vector, $\bar{E}$; the Sun unit vector, $\hat{S}$; and the magnetic field vector, $\bar{M}$. The conversion of IR scanner data to $\hat{E}_B$, Sun data to $\hat{S}_B$, and magnetometer data to $\bar{M}_B$ is detailed in Section 4.3.3. The hierarchy of the attitude sensors, with regard to inherent accuracy, is IR scanner (Earth position), Sun sensor (Sun position), and magnetometer (magnetic field vector), in decreasing order. Thus, if the spacecraft attitude were to be determined from IR scanner and magnetometer data, the primary reference vector used in Equations (4-74) through (4-78) would be $\hat{E}$, and the secondary, $\hat{M}$.

It is advantageous to choose the primary reference vector $\hat{a}_B$ from the more accurate sensor because the effect of Equation (4-77) is to force the pitch, roll, and yaw solutions to properly transform $\hat{a}_O$ to $\hat{a}_B$, and to use data from the less accurate sensor only to resolve the phase angle about $\hat{a}_B$.

4.4.1.2 Attitude Determination for Degraded Magnetometer Data

It is possible that the data from one or more of the magnetometer axes could be degraded because of saturation or large uncertainties in the residual magnetometer bias at various stages of the mission. During acquisition mode, a three-axis attitude must be obtained from Sun sensor and the degraded magnetometer data, while in mission mode during night passes, the only data available for attitude determination would be IR scanner and degraded magnetometer data. The following describes an algorithm for computing a three-axis attitude from degraded magnetometer data.

Let $\hat{R}_B = (R_{B1}, R_{B2}, R_{B3})^T$ be the reference unit vector in spacecraft coordinates obtained from Sun sensor $(\hat{R}_B = \hat{S}_B)$ or IR scanner $(\hat{R}_B = \hat{E}_B)$ data and $\hat{R}_I = (R_1, R_2, R_3)^T$ be the same reference vector in inertial coordinates. Let

$$\begin{aligned} \alpha_B &= \tan^{-1} (R_{B2}/R_{B1}) \\ \delta_B &= \sin^{-1} (R_{B3}) \\ \alpha_I &= \tan^{-1} (R_2/R_1) \\ \delta_I &= \sin^{-1} (R_3) \end{aligned} \tag{4-79}$$

A rotation about $\hat{Z}_I$ through the angle $\Delta\alpha = \alpha_I - \alpha_B$ forms a new coordinate system, denoted by the subscript N, in which $\hat{R}_N$ has azimuth α_B (Figure 4-5). A second rotation about the unit vector $\hat{V}_I = (\cos(\alpha_I - \pi/2), \sin(\alpha_I - \pi/2), 0)^T = (\sin\alpha_I, -\cos\alpha_I, 0)^T$ through the angle $\Delta\delta = \delta_I - \delta_B$ creates a new coordinate system, denoted by the subscript M, in which the elevation of $\hat{R}_M$ is δ_B (Figure 4-6). Because the reference vector $\hat{R}$ has azimuth α_B and elevation δ_B in both the M and the spacecraft coordinate systems, the (X_M, Y_M, Z_M) coordinate system is the same as the spacecraft coordinate system except for a rotation about R_I. In GI coordinates, the basis vectors of the N- reference axes are

$$\begin{aligned}
\hat{X}_N &= (\cos\Delta\alpha, \sin\Delta\alpha, 0)^T \\
\hat{Y}_N &= (-\sin\Delta\alpha, \cos\Delta\alpha, 0)^T \\
\hat{Z}_N &= (0, 0, 1)^T
\end{aligned} \tag{4-80}$$

An application of Equation (4-5), with $\theta = \Delta\delta$, $a = \hat{V}_I$, and $b = \hat{X}_N$, $\hat{Y}_N$ and $\hat{Z}_N$, yields

$$\hat{X}_M = \begin{pmatrix} X_{M1} \\ X_{M2} \\ X_{M3} \end{pmatrix} = \begin{pmatrix} \sin\alpha_I \sin\alpha_B + \cos\Delta\delta\,(\cos\Delta\alpha - \sin\alpha_I \sin\alpha_B) \\ -\sin\alpha_B \cos\alpha_I + \cos\Delta\delta\,(\sin\Delta\alpha + \sin\alpha_B \cos\alpha_I) \\ \sin\Delta\delta \cos\alpha_B \end{pmatrix} \tag{4-81a}$$

$$\hat{Y}_M = \begin{pmatrix} Y_{M1} \\ Y_{M2} \\ Y_{M3} \end{pmatrix} = \begin{pmatrix} -\cos\alpha_B \sin\alpha_I + \cos\Delta\delta\,(-\sin\Delta\alpha + \cos\alpha_B \sin\alpha_I) \\ \cos\alpha_B \cos\alpha_I + \cos\Delta\delta\,(\cos\Delta\alpha - \cos\alpha_B \cos\alpha_I) \\ \sin\Delta\delta \sin\alpha_B \end{pmatrix} \tag{4-81b}$$

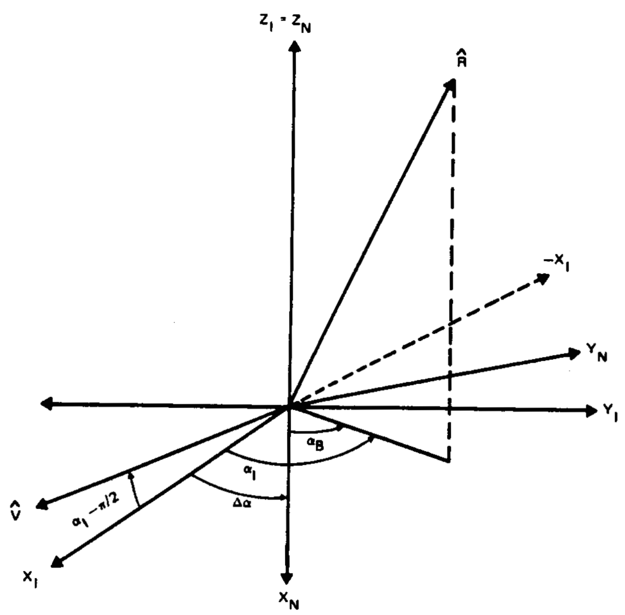


Figure 4-5. Transformation From GI to First Intermediate Coordinate System (N)

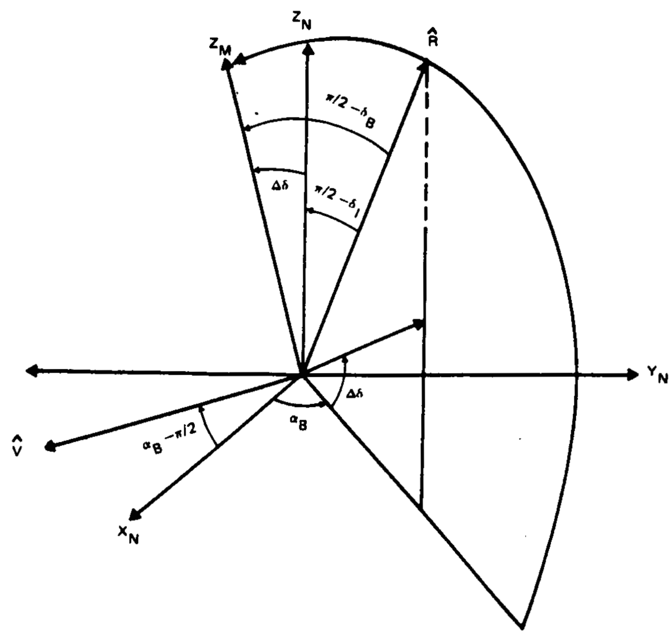


Figure 4-6. Transformation From First Intermediate (N) to Second Intermediate Coordinate System (M)

$$\hat{Z}_M = \begin{pmatrix} Z_{M1} \\ Z_{M2} \\ Z_{M3} \end{pmatrix} = \begin{pmatrix} -\sin \Delta\delta \cos \alpha_I \\ -\sin \Delta\delta \sin \alpha_I \\ \cos \Delta\delta \end{pmatrix} \tag{4-81c}$$

where the basis vectors of the M- reference axes are now expressed in GI coordinates.

The remaining rotation is about the vector $\hat{R}_I$ through an arbitrary angle θ to a new coordinate system denoted by the subscript P . Another application of Equation (4-5), yields the basis vectors of the P- reference axes in GI coordinates.

$$\hat{X}_P = \begin{pmatrix} X_{P1} \\ X_{P2} \\ X_{P3} \end{pmatrix} = \begin{pmatrix} R_1 (\hat{R}_I \cdot \hat{X}_M) + \cos\theta (X_{M1} - R_1 (\hat{R}_I \cdot \hat{X}_M)) + \sin\theta (R_2 X_{M3} - R_3 X_{M2}) \\ R_2 (\hat{R}_I \cdot \hat{X}_M) + \cos\theta (X_{M2} - R_2 (\hat{R}_I \cdot \hat{X}_M)) + \sin\theta (R_3 X_{M1} - R_1 X_{M3}) \\ R_3 (\hat{R}_I \cdot \hat{X}_M) + \cos\theta (X_{M3} - R_3 (\hat{R}_I \cdot \hat{X}_M)) + \sin\theta (R_1 X_{M2} - R_2 X_{M1}) \end{pmatrix} \tag{4-82a}$$

$$\hat{Y}_P = \begin{pmatrix} Y_{P1} \\ Y_{P2} \\ Y_{P3} \end{pmatrix} = \begin{pmatrix} R_1 (\hat{R}_I \cdot \hat{Y}_M) + \cos\theta (Y_{M1} - R_1 (\hat{R}_I \cdot \hat{Y}_M)) + \sin\theta (R_2 Y_{M3} - R_3 Y_{M2}) \\ R_2 (\hat{R}_I \cdot \hat{Y}_M) + \cos\theta (Y_{M2} - R_2 (\hat{R}_I \cdot \hat{Y}_M)) + \sin\theta (R_3 Y_{M1} - R_1 Y_{M3}) \\ R_3 (\hat{R}_I \cdot \hat{Y}_M) + \cos\theta (Y_{M3} - R_3 (\hat{R}_I \cdot \hat{Y}_M)) + \sin\theta (R_1 Y_{M2} - R_2 Y_{M1}) \end{pmatrix} \tag{4-82b}$$

$$\hat{Z}_P = \begin{pmatrix} Z_{P1} \\ Z_{P2} \\ Z_{P3} \end{pmatrix} = \begin{pmatrix} R_1 (\hat{R}_I \cdot \hat{Z}_M) + \cos\theta (Z_{M1} - R_1 (\hat{R}_I \cdot \hat{Z}_M)) + \sin\theta (R_2 Z_{M3} - R_3 Z_{M2}) \\ R_2 (\hat{R}_I \cdot \hat{Z}_M) + \cos\theta (Z_{M2} - R_2 (\hat{R}_I \cdot \hat{Z}_M)) + \sin\theta (R_3 Z_{M1} - R_1 Z_{M3}) \\ R_3 (\hat{R}_I \cdot \hat{Z}_M) + \cos\theta (Z_{M3} - R_3 (\hat{R}_I \cdot \hat{Z}_M)) + \sin\theta (R_1 Z_{M2} - R_2 Z_{M1}) \end{pmatrix} \tag{4-82c}$$

where $\hat{R}_I = (R_1, R_2, R_3)^T$.

$[T(\theta)]$, which defines the transformation from GI coordinates to the coordinate system denoted by the subscript P, is given by

$$[T(\theta)] = \begin{bmatrix} \hat{X}_P^T \\ \hline \hat{Y}_P^T \\ \hline \hat{Z}_P^T \end{bmatrix} \tag{4-83}$$

For some rotation angle θ_A about $\hat{R}_I$, the P and the spacecraft coordinate systems are identical. For this rotation angle, $[A] = [T(\theta_A)]$, where $[A]$ is the attitude matrix. Thus, to determine the attitude, it is sufficient to determine the correct rotation angle θ_A.

$[T(\theta_A)]$ may be rewritten as

$$[T(\theta_A)] = \left[\begin{array}{c|c|c} a_{11}\cos\theta_A + b_{11}\sin\theta_A + c_{11} & a_{12}\cos\theta_A + b_{12}\sin\theta_A + c_{12} & a_{13}\cos\theta_A + b_{13}\sin\theta_A + c_{13} \\ a_{21}\cos\theta_A + b_{21}\sin\theta_A + c_{21} & a_{22}\cos\theta_A + b_{22}\sin\theta_A + c_{22} & a_{23}\cos\theta_A + b_{23}\sin\theta_A + c_{23} \\ a_{31}\cos\theta_A + b_{31}\sin\theta_A + c_{31} & a_{32}\cos\theta_A + b_{32}\sin\theta_A + c_{32} & a_{33}\cos\theta_A + b_{33}\sin\theta_A + c_{33} \end{array}\right] \tag{4-84}$$

where a_{ij}, b_{ij}, and c_{ij} are determined from Equation (4-82) with $\theta \equiv \theta_A$. In particular, $[T(\theta_A)]$ satisfies

$$[T(\theta_A)]\ \bar{M}_I = \bar{M}_B \tag{4-85}$$

where $\bar{M}_I$ is the magnetic field vector in inertial coordinates and $\bar{M}_B$ is the observed magnetic field vector in body coordinates after correcting for all biases. Substitution of Equation (4-84) into Equation (4-85) yields

$$
\begin{aligned}
k_{11} \cos \theta_A + k_{12} \sin \theta_A + k_{13} &= M_{B1} \\
k_{21} \cos \theta_A + k_{22} \sin \theta_A + k_{23} &= M_{B2} \\
k_{31} \cos \theta_A + k_{32} \sin \theta_A + k_{33} &= M_{B3}
\end{aligned}
\tag{4-86}
$$

where

$$
\begin{aligned}
k_{i1} &= a_{i1} M_{I1} + a_{i2} M_{I2} + a_{i3} M_{I3} \\
k_{i2} &= b_{i1} M_{I1} + b_{i2} M_{I2} + b_{i3} M_{I3} \\
k_{i3} &= c_{i1} M_{I1} + c_{i2} M_{I2} + c_{i3} M_{I3}
\end{aligned}
\tag{4-87}
$$

for $i = 1, 2, 3$.

Any two components of the observed magnetic field vector are sufficient to determine θ_A. For example, suppose that the ith and jth magnetometer components provide valid data. Then, from Equation (4-86) and (4-87),

$$
\frac{k_{i1} \cos \theta_A + k_{i2} \sin \theta_A}{k_{j1} \cos \theta_A + k_{j2} \sin \theta_A} = \frac{M_{Bi} - k_{i3}}{M_{Bj} - k_{j3}}
\tag{4-88}
$$

from which

$$\theta_A = \mathrm{ATAN2}\left[\frac{k_{j1}\,(M_{Bi} - k_{i3}) - k_{i1}\,(M_{Bj} - k_{j3})}{k_{i2}\,(M_{Bj} - k_{j3}) - k_{j2}\,(M_{Bi} - k_{i3})}\right] \tag{4-89}$$

When [A] is thus determined, [K] is given by

$$[K] = [A]\,[R]^T \tag{4-90}$$

Pitch, roll, and yaw angles are then obtained from Equation (4-4).

It is sometimes possible to determine θ_A from only one magnetometer component denoted by i . In this case, Equation (4-86) may be rewritten as

$$-k_{i1} \cos \theta_A = k_{i2} \sin \theta_A + k_{i3} - M_{Bi} \tag{4-91}$$

To solve this equation, set

$$\begin{aligned} d &= \left(k_{i1}^2 + k_{i2}^2\right)^{1/2} \\ a &= k_{i1}/d \\ b &= k_{i2}/d \\ c &= -(k_{i3} - M_{Bi})/d \end{aligned} \tag{4-92}$$

If $d = 0$, then any value of θ_A solves Equation (4-91), which means that the ith component of the magnetometer data yields no further attitude information.

However, if $d \neq 0$, a and b are well defined and satisfy $a^2 + b^2 = 1$, so that w may be chosen such that

$$a = \cos w \tag{4-93a}$$

$$b = \sin w \tag{4-93b}$$

where

$$w = \text{ATAN2}\ (b,\ a) \tag{4-93c}$$

Equation (4-91) can now be written

$$\cos w \cos \theta_A + \sin w \sin \theta_A = c \tag{4-94a}$$

or

$$\cos (\theta_A - w) = c \tag{4-94b}$$

which has solutions

$$\theta_A = w \pm \cos^{-1} (c) = \text{ATAN2}\ (b,\ a) \pm \cos^{-1} (c) \tag{4-95}$$

After the two possible solutions θ_A and θ'_A to Equation (4-91) have been found, it remains to determine the correct solution. In mission mode, this can be done by solving for $[K(\theta_A)]$ and $[K(\theta'_A)]$ in Equation (4-90) and computing pitch, roll, and yaw angles p_A, r_A, y_A and p'_A, r'_A, y'_A for θ_A and θ'_A, respectively. The correct angle is chosen by comparing the two pitch, roll, and yaw solutions to the null attitude, and selecting the angle which generates the attitude closest to null.

4.4.2 Magnetometer Bias Determination

The magnetometers flown on HCMM are subject to residual biases from a variety of sources: uncompensated spacecraft residual magnetic dipoles, uncompensated effects from control electromagnet activity, and operation of the HCMR. Some of these sources, such as the spacecraft residual magnetic dipoles, will produce long-term, relatively stable biases, while other sources produce short-term, time-dependent biases. For this reason, the magnetometer biases must be redetermined at frequent intervals. Because of the low data volume (an average pass is only 10 minutes long), filtering techniques which require a large data base, such as RESIDU, are unacceptable. However, it is possible to obtain an estimate of the residual biases deterministically when IR scanner and Sun sensor data are available.

At times $t_1, t_2, \ldots, t_N$, assume that the following sensor and ephemeris data is available:

- IR scanner p' and r' (with biases and systematic errors due to oblateness, CO_2 layer variations and eccentricity removed)
- Sun sensor N_a, N_b, and $\underline{ID}$
- Magnetometer triad reading, $\bar{M}_S$
- Inertial Sun unit vector, $\hat{S}_I$
- Inertial magnetic field vector, $\bar{M}_I$
- Earth to spacecraft vector, $\bar{E}_I$
- Inertial to orbital transformation, [R]

At each time t_i, the IR scanner data determines the Earth to spacecraft unit vector in spacecraft coordinates, $\bar{E}_B$, from Equation (4-46) and the Sun sensor data defines the Sun unit vector in spacecraft coordinates, $\hat{S}_B$, from Equation (4-53). The measured magnetic field in spacecraft coordinates, $\bar{M}_B$, before compensating for residual biases, is

$$\bar{M}_B = [M]\,\bar{M}_S \tag{4-96}$$

where $[M]$ is the matrix of direction cosines which define the orientation of the magnetometer triad spacecraft coordinates.

The orbital to spacecraft transformation $[K]$ can be determined from IR scanner and Sun sensor data using the nominal attitude determination method given in Equations (4-74) through (4-78), where $\hat{E}_B$ is chosen to be the primary reference vector. The attitude matrix $[A]$ is then

$$[A] = [K]\,[R] \tag{4-97}$$

Define $\bar{M}'_B$, the magnetic field in spacecraft coordinates predicted from the inertial field and the attitude matrix, by

$$\bar{M}'_B = [A]\,\bar{M}_I \tag{4-98}$$

Hence, the resulting estimates of the residual bias $\bar{r}$ are

$$\bar{r} = \bar{M}'_B - \bar{M}_B \tag{4-99}$$

Denoting the estimate of the bias at time t_j by $\bar{r}_j$, the average bias $\bar{r}_{AV}$ is

$$\bar{r}_{AV} = \frac{1}{N}\sum_{j=1}^{N} \bar{r}_j \tag{4-100}$$

The estimated error in the ith component of $\bar{r}$, denoted $\Delta\bar{r}_i$, is

$$\Delta\bar{r}_i = \frac{1}{N-1}\left[\sum_{j=1}^{N}\left(\bar{r}_j^i - \bar{r}_{AV}^i\right)^2\right]^{1/2}, \quad i = 1, 2, 3 \tag{4-101}$$

where $\bar{r}_j^i$ denotes the ith component of the jth estimate of the bias.

The accuracy of the estimates of the residual magnetometer biases is strongly affected by the quality of the attitude solutions provided by the IR scanner and Sun sensor data. Because the Earth to spacecraft vector has been selected as the primary vector in obtaining the discrete attitude solutions, the Sun sensor data is only used to resolve the yaw angle or phase angle about $\hat{E}$. The best geometry for accurate resolution of the yaw angle occurs when there is a large angular separation between the Sun vector and the orbital yaw axis, i.e., when

$$|\hat{S}_O \cdot \hat{E}_O| = |S_{03}| < \cos t \tag{4-102}$$

where t is a user-specified minimum angular separation. Only Sun sensor data which satisfies this condition should be used for magnetometer bias determination.

4.4.3 Differential Corrector

When attitude is computed using discrete methods, it may be necessary to bridge gaps in the data by interpolation or smoothing to produce attitude solutions at the required times. Large gaps make this method prone to significant errors, and therefore an alternative method of attitude determination is desirable. One alternative is an estimator, or filter, which models attitude propagation as a polynomial function of time. The filter uses sensor data to estimate the values of the coefficients of the polynomial and the uncertainties in this estimation. When good estimates of the polynomial coefficients are known, it is possible to obtain accurate attitude solutions at any time in the interval.

One such filter is based on the Gauss-Newton method which applies a weighted least squares linear estimator to the linearized version of a nonlinear problem. This method (Reference 4) requires definition of the n-dimensional state vector $\bar{x} = (x_1, x_2, \ldots, x_n)^T$, which contains the parameters to be estimated; the m-dimensional observation vector $\bar{y} = (y_1, \ldots, y_m)^T$, and the m-dimensional observation model vector $\bar{z} = (z_1, \ldots, z_m)^T$ which depends on the state vector $\bar{x}$; and a vector of measurements $\bar{m} = (m_1, m_2, \ldots, m_p)^T$. This is expressed

$$\bar{z} = \bar{f}(\bar{x}, \bar{m}) = (f_1(\bar{x}, \bar{m}), f_2(\bar{x}, \bar{m}), \ldots, f_m(\bar{x}, \bar{m}))^T \tag{4-103}$$

For N distinct measurement vectors $\bar{m}^1, \bar{m}^2, \ldots, \bar{m}^N$ and observation vectors $\bar{y}^1, \bar{y}^2, \ldots, \bar{y}^N$ define

$$\bar{\bar{y}} = \left(\bar{y}^1, \bar{y}^2, \ldots, \bar{y}^N\right)^T = \left(y_1^1, y_2^1, \ldots, y_m^1, y_1^2, \ldots, y_m^2, \ldots, y_1^N, \ldots, y_m^N\right)^T \tag{4-104}$$

and

$$\bar{\bar{f}}(\bar{x}, \bar{m}) = [\bar{f}(\bar{x}, \bar{m}^1), \ldots, \bar{f}(\bar{x}, \bar{m}^N)]^T = [f_1(\bar{x}, m^1), \ldots, f_m(\bar{x}, \bar{m}^1), \ldots, f_1(\bar{x}, \bar{m}^N), \ldots, f_m(\bar{x}, \bar{m}^N)]^T \tag{4-105}$$

$$
F(\bar{x}) = \begin{bmatrix}
\frac{\partial f_1(\bar{x}, \bar{m}^1)}{\partial x_1} & \frac{\partial f_1(\bar{x}, \bar{m}^1)}{\partial x_2} & \cdots & \frac{\partial f_1(\bar{x}, \bar{m}^1)}{\partial x_n} \\
\vdots & & & \\
\frac{\partial f_m(\bar{x}, \bar{m}^1)}{\partial x_1} & \frac{\partial f_m(\bar{x}, \bar{m}^1)}{\partial x_2} & \cdot & \frac{\partial f_m(\bar{x}, \bar{m}^1)}{\partial x_n} \\
\vdots & & & \\
\frac{\partial f_1(\bar{x}, \bar{m}^N)}{\partial x_1} & \frac{\partial f_1(\bar{x}, \bar{m}^N)}{\partial x_2} & & \frac{\partial f_1(\bar{x}, \bar{m}^N)}{\partial x_n} \\
\vdots & & & \\
\frac{\partial f_m(\bar{x}, \bar{m}^N)}{\partial x_1} & \frac{\partial f_m(\bar{x}, \bar{m}^N)}{\partial x_2} & & \frac{\partial f_m(\bar{x}, \bar{m}^N)}{\partial x_n}
\end{bmatrix} \qquad (4\text{-}106)
$$

Note that $\bar{\bar{y}}$ and $\bar{\bar{f}}(\bar{x}, \bar{m})$ are vectors of length $m \times N$ and $F(\bar{x})$ is a matrix having $m \times N$ rows and n columns. The iteration scheme is

$$
\bar{x}_{k+1} = \bar{x}_k + [F^T(\bar{x}_k) W^{-1} F(\bar{x}_k)]^{-1} F^T(\bar{x}_k) W^{-1} [\bar{\bar{y}} - \bar{\bar{f}}(\bar{x}_k, \bar{m})] \qquad (4\text{-}107)
$$

where the matrix of observation weights N is given by

$$W = \begin{bmatrix} w_1^1 & & & & & & & & \\ & w_2^1 & & & & & & 0 & \\ & & \ddots & & & & & & \\ & & & w_m^1 & & & & & \\ & & & & w_1^2 & & & & \\ & & & & & \ddots & & & \\ & & & & & & w_1^2 & & \\ & & & & & & & \ddots & \\ & 0 & & & & & & & w_1^N \\ & & & & & & & & & \ddots \\ & & & & & & & & & & w_m^N \end{bmatrix}$$

and w_j^i is the weight attached to the jth component of the ith observation.

When the differential corrector is used for attitude determination, the attitude propagation equation at time t is

$$p(T) = p_0 + p_1 T + p_2 T^2/2 \tag{4-108a}$$

$$r(T) = r_0 + r_1 T + r_2 T^2/2 \tag{4-108b}$$

$$y(T) = y_0 + y_1 T + y_2 T^2/2 \tag{4-108c}$$

where $T = [2t - (t_0 + t_N)]/(t_N - t_0)$

t_0 = start time of processing interval

t_N = stop time of processing interval

$p(T)$ = pitch angle (Euler 2-1-3 rotation)

$r(T)$ = roll angle (Euler 2-1-3 rotation)

$y(T)$ = yaw angle (Euler 2-1-3 rotation)

The state vector is

$$\bar{x} = (p_0, p_1, p_2, r_0, r_1, r_2, y_0, y_1, y_2, b_1, b_2, b_3)^T \qquad (4\text{-}109)$$

where $\bar{b} = (b_1, b_2, b_3)^T$ is the magnetometer residual bias vector.

The observation and observation model vectors are defined by the following considerations. Estimated values of pitch, roll, and yaw at time t given by Equation (4-108) provide an estimate of the orbital to spacecraft transformation $[K]$ from Equation (4-3). IR scanner, Sun sensor, and magnetometer data at time t provide a measurement of the Earth to spacecraft and Sun unit vectors, and the magnetic field vector in body coordinates. Denote these vectors by $\hat{E}_B$, $\hat{S}_B$, and $\bar{M}_B$, respectively. The same vectors in inertial coordinates, $\hat{E}_I$, $\hat{S}_I$, and $\bar{M}_I$ at time t are obtained from ephemeris and a magnetic field model. Estimated values of the reference vectors in body coordinates (denoted by $\underset{\sim}{X}_B$), based on the inertial vectors and the estimated pitch, roll, and yaw, are

$$\underset{\sim}{\hat{E}}_B = [K]\,[R]\,\hat{E}_I \qquad (4\text{-}110a)$$

$$\underset{\sim}{\hat{S}}_B = [K]\,[R]\,\hat{S}_I \qquad (4\text{-}110b)$$

$$\underset{\sim}{\bar{M}}_B = [K]\,[R]\,\bar{M}_I + \bar{b} \qquad (4\text{-}110c)$$

where $[R]$ is the inertial to orbital transformation. Because the Earth to spacecraft unit vector in orbital coordinates, given by $[R]\,\hat{E}_I$, equals $(0,0,-1)^T$, Equation (4-110a) is

$$\underset{\sim}{\hat{E}}_B = (-k_{13}, -k_{23}, -k_{33})^T \qquad (4\text{-}111)$$

The estimated scanner pitch and roll output, $\underset{\sim}{p}'$ and $\underset{\sim}{r}'$, is then

$$\underset{\sim}{p}' = \text{ATAN2}\,(-k_{13}, k_{33}) \tag{4-112a}$$

$$\underset{\sim}{r}' = \sin^{-1}\,(k_{23}) \tag{4-112b}$$

Define the observation and observation model vectors,

$$\bar{y} = (p', r', S_{B1}, S_{B2}, S_{B3}, M_{B1}, M_{B2}, M_{B3})^T \tag{4-113a}$$

$$\bar{z} = f(\bar{x}, \bar{m}) = (\underset{\sim}{p}', \underset{\sim}{r}', \underset{\sim}{S}_{B1}, \underset{\sim}{S}_{B2}, \underset{\sim}{S}_{B3}, \underset{\sim}{M}_{B1}, \underset{\sim}{M}_{B2}, \underset{\sim}{M}_{B3})^T \tag{4-113b}$$

Expanding Equation (4-113), and using the abbreviation sr for sin r(T), cr for cos r(T), etc., yields

$$\begin{aligned}
\underset{\sim}{p}' &= f_1(\bar{x}, \bar{m}) = \text{ATAN2}\,(\text{sp cy} - \text{cp sr sy}, \text{cp cr}) \\
\underset{\sim}{r}' &= f_2(\bar{x}, \bar{m}) = \sin^{-1}\,(\text{sp sy} + \text{cp sr cy}) \\
\underset{\sim}{S}_{B1} &= f_3(\bar{x}, \bar{m}) = (\text{cp cy} + \text{sp sr sy})\, S_{O1} + \text{cr sy}\, S_{O2} \\
&\qquad + (-\text{sp cy} + \text{cp sr sy})\, S_{O3} \\
\underset{\sim}{S}_{B2} &= f_4(\bar{x}, \bar{m}) = (-\text{cp sy} + \text{sp sr cy})\, S_{O1} + \text{cr cy}\, S_{O2} \\
&\qquad + (\text{sp sy} + \text{cp sr cy})\, S_{O3}
\end{aligned} \tag{4-114}$$

$$\underset{\sim}{S}_{B3} = f_5(\bar{x}, \bar{m}) = sp\ cr\ S_{O1} - sr\ S_{O2} + cp\ cr\ S_{O3}$$

$$\underset{\sim}{M}_{B1} = f_6(\bar{x}, \bar{m}) = (cp\ cy + sp\ sr\ sy)\ M_{O1} + cr\ sy\ M_{O2} + (-sp\ cy + cp\ sr\ sy)\ M_{O3} + b_1$$

$$\underset{\sim}{M}_{B2} = f_7(\bar{x}, \bar{m}) = (-cp\ sy + sp\ sr\ cy)\ M_{O1} + cr\ cy\ M_{O2} + (sp\ sy + cp\ sr\ cy)\ M_{O3} + b_2 \qquad \text{(4-114) (Cont'd)}$$

$$\underset{\sim}{M}_{B3} = f_8(\bar{x}, \bar{m}) = sp\ cr\ M_{O1} - sr\ M_{O2} + cp\ cr\ M_{O3} + b_3$$

From this information, the matrix of partial derivatives may be computed. These derivations are found in Appendix A.

Because the observation vector $\bar{y}$ consists of sensor measurements, the values of the weights in the weight matrix [W] may be specified. In general,

$$W_j^i = \frac{1}{\left(\sigma_j^i\right)^2} \qquad \text{(4-115)}$$

where σ_j^i is the standard deviation of the jth component of the ith sensor measurement. If the data corresponding to this component is not available, set $W_j^i = 0$.

Equation (4-108) may be differentiated to obtain estimates of the pitch, roll, and yaw rates.

$$\dot{p}\ (t) = 2\ (p_1 + p_2\ T)/(t_N - t_0) \qquad \text{(4-116a)}$$

$$\dot{r}\ (t) = 2\ (r_1 + r_2\ T)/(t_N - t_0) \qquad \text{(4-116b)}$$

$$\dot{y}(t) = 2(y_1 + y_2 T)/(t_N - t_0) \tag{4-116c}$$

The mean and standard deviation of the residuals,

$$\epsilon_m = \sum_{i=1}^{N} \sum_{j=1}^{m} \left[y_j^i - f_j\left(\bar{x}_k, \bar{m}^i\right) \right] \Big/ m \cdot N \tag{4-117}$$

$$\epsilon_{SD}^2 = \frac{1}{m \cdot N - 1} \sum_{i=1}^{N} \sum_{j=1}^{m} \left[y_j^i - f_j\left(\bar{x}_k, \bar{m}^i\right) \right]^2 \tag{4-118}$$

as calculated at the $k + 1$st step, based on the previous estimate $\bar{x}_k$, of the state. These values may be used to evaluate the convergence of the algorithm. The normal matrix [N] at the kth iteration is the $n \times n$ matrix defined by

$$\left[N\right] = \left[F^T\left(\bar{x}_k\right) W^{-1} F\left(\bar{x}_k\right)\right] \tag{4-119}$$

Its inverse [C] is called the covariance matrix, and the standard deviation σ_i of the ith element of the state vector is

$$\sigma_i = \left(c_{ii}\right)^{1/2} \tag{4-120}$$

provided (Reference 5):

1. The measurement noise is unbiased.

2. The elements of the weight matrix [W] are known accurately.

3. The mathematical model of the attitude (Equation (4-98)) reflects the attitude propagation exactly.

The above criteria can never be met precisely, and the degree to which these criteria are violated determines how accurately Equation (4-120) estimates the uncertainty of the state vector elements.

The uncertainties of the pitch, roll, and yaw angles evaluated at time T are

$$\sigma_p = \left(\sigma_1^2 + T^2 \sigma_2^2 + T^4 \sigma_3^2 / 4\right)^{1/2} \tag{4-121a}$$

$$\sigma_r = \left(\sigma_4^2 + T^2 \sigma_5^2 + T^4 \sigma_6^2 / 4\right)^{1/2} \tag{4-121b}$$

$$\sigma_y = \left(\sigma_7^2 + T^2 \sigma_8^2 + T^4 \sigma_9^2 / 4\right)^{1/2} \tag{4-121c}$$

4.4.4 Attitude Uncertainty Determination

The accuracy of discrete attitude solutions is degraded by errors in the data used to obtain the solutions. These errors include analog-to-digital conversion of magnetometer and scanwheel data, quantization of Sun sensor data, errors in inertial reference vectors, and uncertainties in the pitch, roll, and magnetometers biases. These errors may be compounded by the basis geometry for the attitude solution and the particular method of attitude determination.

Let $m_1, m_2, \ldots, m_k$ be quantities involved in the computation of pitch, roll, and yaw which are subject to errors $\delta m_1, \delta m_2, \ldots, \delta m_k$. This is expressed as

$$p = p(m_1, m_2, \ldots, m_k) \tag{4-122a}$$

$$r = r(m_1, m_2, \ldots, m_k) \tag{4-122b}$$

$$y = y(m_1, m_2, \ldots, m_k) \tag{4-122c}$$

The errors in pitch, roll, and yaw due to errors in m_1, m_2, ..., m_k can be approximated by the first order terms of a Taylor series,

$$\delta p = \sum_{n=1}^{k} \frac{\partial p}{\partial m_n} \delta m_n \tag{4-123a}$$

$$\delta r = \sum_{n=1}^{k} \frac{\partial r}{\partial m_n} \delta m_n \tag{4-123b}$$

$$\delta y = \sum_{n=1}^{k} \frac{\partial y}{\partial m_n} \delta m_n \tag{4-123c}$$

In matrix notation

$$\begin{pmatrix} \delta p \\ \delta r \\ \delta y \end{pmatrix} = [P] \begin{pmatrix} \delta m_1 \\ \delta m_2 \\ \cdot \\ \cdot \\ \cdot \\ \delta m_k \end{pmatrix} \tag{4-124}$$

where $[P]$ is the $3 \times k$ sensitivity matrix of partial derivatives

$$[P] = \left[\frac{\delta(p, r, y)}{\delta(m_1, m_2, \ldots, m_k)} \right]$$

The covariance matrix defining the attitude uncertainties

$$[\Lambda_c] = \begin{bmatrix} \sigma_p^2 & \sigma_{pr} & \sigma_{py} \\ \sigma_{rp} & \sigma_r^2 & \sigma_{ry} \\ \sigma_{yp} & \sigma_{yr} & \sigma_y^2 \end{bmatrix} \tag{4-126}$$

is related to the diagonal measurement covariance matrix

$$[\Lambda_m] = \begin{bmatrix} \sigma_1^2 & & & \\ & \sigma_2^2 & & \\ & & \ddots & \\ & & & \sigma_k^2 \end{bmatrix} \tag{4-127}$$

by

$$[\Lambda_c] = [P]\,[\Lambda_m]\,[P]^T \tag{4-128}$$

where σ_i^2 is the variance of the random variable δm_i .

The diagonal elements of $[\Lambda_c]$, the variances of the pitch, roll, and yaw errors, are

$$\sigma_p^2 = \sum_{n=1}^{k} \left(\frac{\partial p}{\partial m_n}\right)^2 \sigma_n^2 \tag{4-129a}$$

$$\sigma_r^2 = \sum_{n=1}^{k} \left(\frac{\partial r}{\partial m_n}\right)^2 \sigma_n^2 \qquad (4\text{-}129\text{b})$$

$$\sigma_y^2 = \sum_{n=1}^{k} \left(\frac{\partial y}{\partial m_n}\right)^2 \sigma_n^2 \qquad (4\text{-}129\text{c})$$

The off-diagonal elements, such as

$$\sigma_{pr} = \sum_{n=1}^{k} \frac{\partial p}{\partial m_n} \frac{\partial r}{\partial m_n} \sigma_n^2 \qquad (4\text{-}129\text{d})$$

are generally sums with both positive and negative terms, which will tend to cancel. Therefore, it is natural to consider $[\Lambda_c]$ a diagonal matrix, particularly if the number of observables is large.

A method to associate a confidence level to each attitude set (p, r, y) can be derived by first diagonalizing the covariance matrix

$$[\Lambda_c]' = [T]^T [\Lambda_c] [T] = \begin{bmatrix} \sigma'^2_p & 0 & 0 \\ 0 & \sigma'^2_r & 0 \\ 0 & 0 & \sigma'^2_y \end{bmatrix} \qquad (4\text{-}130)$$

An error ellipsoid with semimajor axes proportional to σ'_p, σ'_r, and σ'_y defines an error volume that is a meaningful measure of total attitude error.

The probability that an attitude error is contained in the ellipsoid defined by

$$\left(\frac{x'}{\sigma'_p}\right)^2 + \left(\frac{y'}{\sigma'_r}\right)^2 + \left(\frac{z'}{\sigma'_y}\right)^2 = K^2 \tag{4-131}$$

is the probability that the chi-square random variable for 3 degrees of freedom is less than K^2. The attitude error, ρ, is defined as the radius of a sphere whose volume equals that of the error ellipsoid. Thus,

$$\frac{4\pi}{3} K\sigma'_p \, K\sigma'_r \, K\sigma'_y = \frac{4}{3}\pi\rho^3 \tag{4-132}$$

and

$$\begin{aligned} \rho = K\left(\sigma'_p \, \sigma'_r \, \sigma'_y\right)^{1/3} &= K\left|[\Lambda'_c]\right|^{1/6} \\ &= K\left|[\Lambda_c]\right|^{1/6} \end{aligned} \tag{4-133}$$

The attitude error is directly related to the determinant of the covariance matrix. Table 4-2 summarizes the error procedure.

Table 4-2. Error Procedure

K	PROBABILITY THAT ATTITUDE ERROR IS LESS THAN ρ
1.0	0.198
1.5	0.48
1.85	0.68
2.0	0.739
3.0	0.971

The quantities used in the attitude calculations which are subject to error are the magnetometer data M_{S1} , M_{S2} , M_{S3} ; the IR scanner pitch, p' , and roll angle, r' ; Sun sensor reticle counts, N_a and N_b ; the inertial magnetic field M_{I1} , M_{I2} , M_{I3} ; the scanner pitch and roll biases p_B and r_B ; and the magnetometer biases b_1 , b_2 , and b_3 .

For sensor data which has a telemetry bucket size of B such that all analog data in the same bucket $[x_0, x_0 + B]$ is converted to the mean value $x_0 + B/2$ in telemetry, the variance in the errors is $B^2/12$. If, however, analog data in the same bucket is converted to the extreme value x_0 or $x_0 + B$, the variance in the errors is $B^2/3$. This consideration applies to the magnetometer, IR scanner, and Sun sensor data. The uncertainties of the scanner and magnetometer biases are available from the scanner biases and magnetometers biases data sets. Errors in the inertial magnetic field predicted from the IGRF model are typically 2 milligauss in each axis, with a maximum-error of 10 milligauss (Reference 6).

Because of the numerous, and sometimes involved, methods of attitude determination, analytical expressions for the matrix of partial derivatives $[P]$ would be very cumbersome in many cases. For this reason, it is advantageous to obtain these partial derivatives numerically. If m_i is a sensor data type, the partial derivatives of the attitude with respect to m_i may be found by first calculating

$$p_1 = p(m_1, m_2, \ldots, m_{i-1}, m_i + \Delta m_i, m_{i+1}, \ldots, m_k) \qquad (4\text{-}134a)$$

$$r_1 = r(m_1, m_2, \ldots, m_{i-1}, m_i + \Delta m_i, m_{i+1}, \ldots, m_k) \qquad (4\text{-}134b)$$

$$y_1 = y(m_1, m_2, \ldots, m_{i-1}, m_i + \Delta m_i, m_{i+1}, \ldots, m_k) \qquad (4\text{-}134c)$$

where Δm_i is the bucket size of the data type and the method of attitude determination is the same used in obtaining solutions in Equation (4-122).

Then the numeric partial derivatives are:

$$\frac{\partial p}{\partial m_i} = \frac{p_1 - p}{\Delta m_i} \tag{4-135a}$$

$$\frac{\partial r}{\partial m_i} = \frac{r_1 - r}{\Delta m_i} \tag{4-135b}$$

$$\frac{\partial y}{\partial m_i} = \frac{y_1 - y}{\Delta m_i} \tag{4-135c}$$

where p , r , and y are the attitude solutions. Because variations in the estimates of the magnetometer biases affect attitude solutions in precisely the same way as variations in the magnetometer data, the corresponding partial derivatives are the same. For the pitch and roll biases, Equations (4-134) and (4-135) should be evaluated with $\Delta m_i \approx 0.5$ degree. For the inertial magnetic field, these evaluations should use the bucket size of the magnetometer data, $\Delta m_i \approx 4.68$ millioersteds.

4.4.5 Combining Attitude Solution

If discrete methods have been used to compute, several different attitude solutions based on different data combinations will have been computed for the same time. In this case, it is necessary to combine these solutions to obtain an average solution at each time.

Suppose pitch solutions p_1, p_2, --, p_N, with associated standard deviations (Equation (4-129) σ_{p1}, σ_{p2}, --, σ_{pN}, have been computed for a particular time. The weighted average pitch angle p is

$$p = \frac{\sum_{\ell=1}^{N} w_{p\ell}\, p_\ell}{\sum_{\ell=1}^{N} w_{p\ell}} \tag{4-136}$$

where the weight, $w_{p\ell}$, is either equal to the reciprocal of the associated standard deviation or user-specified. The variance of the weighted average solution, determined from Equation (4-129) is

$$\begin{aligned}
\sigma_p^2 &= \sum_{n=1}^{k} \left(\frac{\partial p}{\partial m_n}\right)^2 \sigma_n^2 = \left[\sum_{\ell=1}^{N} w_{p\ell}\right]^{-2} \sum_{n=1}^{k} \left(\sum_{\ell=1}^{N} w_{p\ell} \frac{\partial p_\ell}{\partial m_n}\right)^2 \sigma_n^2 \\
&= \left[\sum_{\ell=1}^{N} w_{p\ell}\right]^{-2} \left[w_{p1}^2 \sum_{n=1}^{k} \left(\frac{\partial p_1}{\partial m_n}\right)^2 \sigma_n^2 + \ldots \right. \\
&\quad + w_{pN}^2 \sum_{n=1}^{k} \left(\frac{\partial p_N}{\partial m_n}\right)^2 \sigma_n^2 + 2\, w_{p1}\, w_{p2} \sum_{n=1}^{k} \frac{\partial p_1}{\partial m_n} \frac{\partial p_2}{\partial m_n} \sigma_n^2 \\
&\quad + 2\, w_{p1}\, w_{p3} \sum_{n=1}^{k} \frac{\partial p_1}{\partial m_n} \frac{\partial p_3}{\partial m_n} \sigma_n^2 + \ldots \\
&\quad \left. + 2\, w_{pN-1}\, w_{pN} \sum_{n=1}^{k} \frac{\partial p_{N-1}}{\partial m_n} \frac{\partial p_N}{\partial m_n} \sigma_n^2 \right]
\end{aligned} \tag{4-137}$$

However,

$$w_{pi}^2 \sum_{n=1}^{k} \left(\frac{\partial p_i}{\partial m_n}\right)^2 \sigma_n^2 = w_{pi}^2 \, \sigma_{pi}^2 \tag{4-138}$$

and if $w_{pi} = (\sigma_{pi})^{-1}$, terms of this form are unity. In any case,

$$\begin{aligned}\sigma_p^2 &= \left[\sum_{\ell=1}^{N} w_{p\ell}\right]^{-2} \left[\sum_{i=1}^{N} w_{pi}^2 \, \sigma_{pi}^2 + 2 \sum_{m=1}^{N-1} \sum_{n=m+1}^{N} w_{pm} \, w_{pn} \sum_{i=1}^{k} \frac{\partial p_m}{\partial m_i} \frac{\partial p_n}{\partial m_i} \sigma_i^2\right] \\ &= \left[\sum_{\ell=1}^{N} w_{p\ell}\right]^{-2} \left[\sum_{i=1}^{N} w_{pi}^2 \, \sigma_{pi}^2 + 2 \sum_{m=1}^{N-1} \sum_{n=M+1}^{N} w_{pm} \, w_{pn} \, \sigma_{pmn}\right]\end{aligned} \tag{4-139}$$

where σ_{pmn} denotes the covariance of the mth and nth solutions for the pitch angle.

In practice, the maximum number of pitch angle solutions calculated using discrete methods is three (i.e., $N \leq 3$). If three solutions have been computed, any two of them must have a sensor data type in common, because each solution is obtained from two of the three available sensor data types. This means that no pair of solutions is statistically independent, so that the covariance terms in Equation (4-139) cannot be neglected. The uncertainty of the weighted average pitch solution is

$$\sigma_p = \left[\sum_{\ell=1}^{N} w_{p\ell}\right]^{-1} \left[\sum_{i=1}^{N} w_{pi}^2 \, \sigma_{pi}^2 + 2 \sum_{m=1}^{N-1} \sum_{n=m+1}^{N} w_{pm} \, w_{pn} \, \sigma_{pmn}\right]^{1/2} \tag{4-140}$$

The weighted average roll and yaw angles and associated uncertainties can be obtained in a completely analogous manner.

$$r = \frac{\sum_{\ell=1}^{N} w_{r\ell}\, r_{\ell}}{\sum_{\ell=1}^{N} w_{r\ell}} \tag{4-141a}$$

$$\sigma_r = \left[\sum_{\ell=1}^{N} w_{r\ell}\right]^{-1} \left[\sum_{i=1}^{N} w_{ri}^2\, \sigma_{ri}^2 + 2 \sum_{m=1}^{N-1} \sum_{n=m+1}^{N} w_{rm}\, w_{rn}\, \sigma_{rmn}\right]^{1/2} \tag{4-141b}$$

where $r_1, r_2, \ldots, r_N$ are the discrete attitude roll angle solutions with standard deviations $\sigma_{r1}, \ldots, \sigma_{rN}$, $w_{rk} = (\sigma_{rk})^{-1}$ or a user-specified value, and σ_{rmn} is the covariance of the mth and nth roll angle solutions. Similarly, for yaw

$$y = \frac{\sum_{\ell=1}^{N} w_{y\ell}\, y_{\ell}}{\sum_{\ell=1}^{N} w_{y\ell}} \tag{4-142a}$$

$$\sigma_y = \left[\sum_{\ell=1}^{N} w_{y\ell}\right]^{-1} \left[\sum_{i=1}^{N} w_{yi}^2\, \sigma_{yi}^2 + 2 \sum_{m=1}^{N-1} \sum_{n=m+1}^{N} w_{ym}\, w_{yn}\, \sigma_{ymn}\right]^{1/2} \tag{4-142b}$$

4.4.6 Smoothing and Attitude Rate Determination

After the discrete attitude solutions have been combined to obtain the weighted average of the pitch, roll, and yaw angles, the average attitude solutions may be smoothed. Smoothing is an effective means of reducing the effect of zero-mean sources of error (such as the quantization of Sun sensor data, digitization of magnetometer and IR scanner data, and errors in the inertial magnetic field model) on the attitude solutions. Also, smoothing is required if there are gaps in the attitude solutions or if attitude rate solutions are desired.

The pitch, roll, and yaw angles are each fit to a Chebyshev polynomial. A general discussion of Chebyshev polynomials can be found in Section 4.3.5. The degree, r, of the fitting polynomial is defined to be the smallest number satisfying

$$S_r < \epsilon S_0 \tag{4-143}$$

where S_r is a measure of the goodness-of-fit of the Chebyshev polynomial of degree r and ϵ is a constant, varying typically between 10^{-3} and 10^{-5}. Also, r must be less than a user-specified maximum degree. In on-orbit mode, this maximum degree should be reasonably small (approximately 3) because under nominal circumstances (steady state control performance, no wheel desaturation), the pitch angle will remain nearly constant and the large time constants of the precession/nutation control result in slow variations in the roll and yaw angles. In acquisition mode, this maximum degree must be larger to accomodate the more rapidly varying attitude solutions.

Attitude rates may be determined from the smoothing polynomials as well. The derivatives of the Chebyshev polynomials satisfy the recursion relation (Reference 1)

$$\dot{g}_0(T) = 0 \tag{4-144a}$$

$$\dot{g}_k(T) = -\frac{k}{(1 - T^2)} \left(Tg_k(T) - g_{k-1}(T)\right) \qquad |T| < 1 \tag{4-144b}$$

$$\dot{g}_k(-1) = \begin{Bmatrix} k^2 & k \text{ odd} \\ -k^2 & k \text{ even} \end{Bmatrix} \tag{4-144c}$$

$$\dot{g}_k(1) = k^2 \tag{4-144d}$$

Thus, if the smoothed estimate of the pitch angle $\overline{p}$, as a function of the scaled time T (see Equation (4-67)), is

$$\overline{p}(T) = \sum_{k=0}^{N} c_k \, g_k(T) \tag{4-145}$$

the pitch rate, as a function of time t, based on the smoothed estimate is

$$\dot{\overline{p}}(t) = C_t \sum_{k=0}^{N} c_k \, \dot{g}_k(T)$$

$$= \frac{C_t}{(1 - T^2)} \left[c_1 + \sum_{k=1}^{N-1} \{(k+1)\, c_{k+1} - kT\, c_k\}\, g_k(T) + Nc_N T\, g_N(T) \right] \tag{4-146a}$$

for $|T| < 1$

$$\dot{\overline{p}}(t_o) = C_t \sum_{k=0}^{N} (-1)^{k+1} c_k \, k^2 \tag{4-146b}$$

$$\dot{\overline{p}}(t_n) = C_t \sum_{k=0}^{N} c_k \, k^2 \tag{4-146c}$$

where t_o and t_n are the minimum and maximum times and $C_t = 2/(t_n - t_o)$. Similar expressions define the roll and yaw rates.

4.4.7 Computation of Monitor and Display Parameters

During acquisition mode, attitude rate data is required to evaluate the effectiveness of the onboard control loop. The following algorithm computes a value for all three components of the spacecraft angular velocity but requires three-axis attitude solutions to do so. After discrete attitude solutions have been obtained for the pass, the pitch, roll, and yaw solutions are smoothed and the pitch, roll, and yaw rates determined as described in Section 4.4.6. The attitude and attitude rate data is related to the angular velocity vector in spacecraft coordinates by

$$\bar{\omega}_B = \begin{pmatrix} \sin y \cos r \, (\dot{p} - \Omega_o) + \dot{r} \cos y \\ \cos r \cos y \, (\dot{p} - \Omega_o) - \dot{r} \sin y \\ \dot{y} - \sin r \, (\dot{p} - \Omega_o) \end{pmatrix} \tag{4-147}$$

where Ω_o is the orbital rate (≈ 0.06 degree per second).

The acquisition control laws drive the spacecraft Y-axis toward the negative orbit normal; therefore, the angle θ_Y between these two vectors should be monitored. The orbit normal unit vector $\hat{n}$ in inertial coordinates is computed from spacecraft ephemeris (Equation (4-1)). The pitch, roll, and yaw angles and ephemeris define the attitude matrix $[A]$ (Equation (4-7)). In inertial coordinates, the Y-axis attitude is

$$\hat{Y}_I = [A]^T \begin{pmatrix} 0 \\ 1 \\ 0 \end{pmatrix} = \begin{pmatrix} a_{21} \\ a_{22} \\ a_{23} \end{pmatrix} \tag{4-148}$$

Therefore

$$\theta_Y = \cos^{-1}(-\hat{n} \cdot \hat{Y}_I) = \cos^{-1}(-n_1 a_{21} - n_2 a_{22} - n_3 a_{23}) \qquad (4\text{-}149)$$

Another important analytic parameter is the angle between the angular velocity vector and the spacecraft angular momentum vector. The total angular momentum is the sum of the angular momentum of the body and that of the scanwheel. In body coordinates,

$$\bar{L}_B = \bar{L}_{BODY} + \bar{L}_{WHEEL} = \overleftrightarrow{I} \cdot \bar{\omega}_B - h_w \hat{j} \qquad (4\text{-}150)$$

where $\overleftrightarrow{I}$ is the moment of inertial tensor and h_w is the scanwheel angular momentum. The magnitude of the total angular momentum is

$$\|\bar{L}_B\| = \left(\bar{L}_B \cdot \bar{L}_B\right)^{1/2} \qquad (4\text{-}151)$$

Therefore,

$$\theta_N = \cos^{-1}\left[\frac{\bar{L}_B \cdot \bar{\omega}_B}{\|\bar{L}_B\| \cdot \|\bar{\omega}_B\|}\right] = \cos^{-1}\left[\frac{(\overleftrightarrow{I}\,\bar{\omega}_B) \cdot \bar{\omega}_B - h_w\,\omega_{B2}}{\|\bar{L}_B\|\ \|\bar{\omega}_B\|}\right] \qquad (4\text{-}152)$$

SECTION 5 - FUNCTIONAL SPECIFICATIONS

This section contains specifications for the functions to be performed by the HCMM Attitude Determination System (ADS). Section 5.1 gives an overview of the requirements of the system; Sections 5.2 through 5.4 contain specifications for the telemetry processor functions, the data adjustment functions, and the attitude determination functions. Section 5.5 specifies the output and logging functions and Section 5.6 describes the data sets required for input and output.

5.1 SYSTEM OVERVIEW

The HCMM ADS requirements may be conveniently divided into four categories. The first category contains the telemetry processor functions, which include reading telemetry data from a disk or tape data set, converting data to engineering units, and providing optional data quality checking. The data adjustment functions include displaying and optionally rejecting outlying data, obtaining ephemeris information, and reducing, by preaveraging or curve fitting, the volume of data required for attitude determination. The attitude determination functions include determining magnetometer biases for sunlit passes in definitive mode and computing the spacecraft attitude. Quality assurance functions are provided, on option, after attitude solutions have been obtained. Finally, the output and logging functions include writing the computed attitude solutions to an attitude history file and logging a summary of the quality and quantity of the processed data and attitude solutions to a permanent log data set.

The flow of data processing is illustrated in Figure 5-1. The raw telemetry data received by the Spaceflight Tracking and Data Network (STDN) tracking station are transmitted via the NASA Communications Network (NASCOM) to the Information Processing Division (IPD) at GSFC and/or to the Operations Control Center (OCC) at GSFC. Normally, data will be directed to the former; only during near-real-time processing will data be received at the OCC. If the NASCOM link is unavailable, raw telemetry tapes will be mailed to IPD (alternate

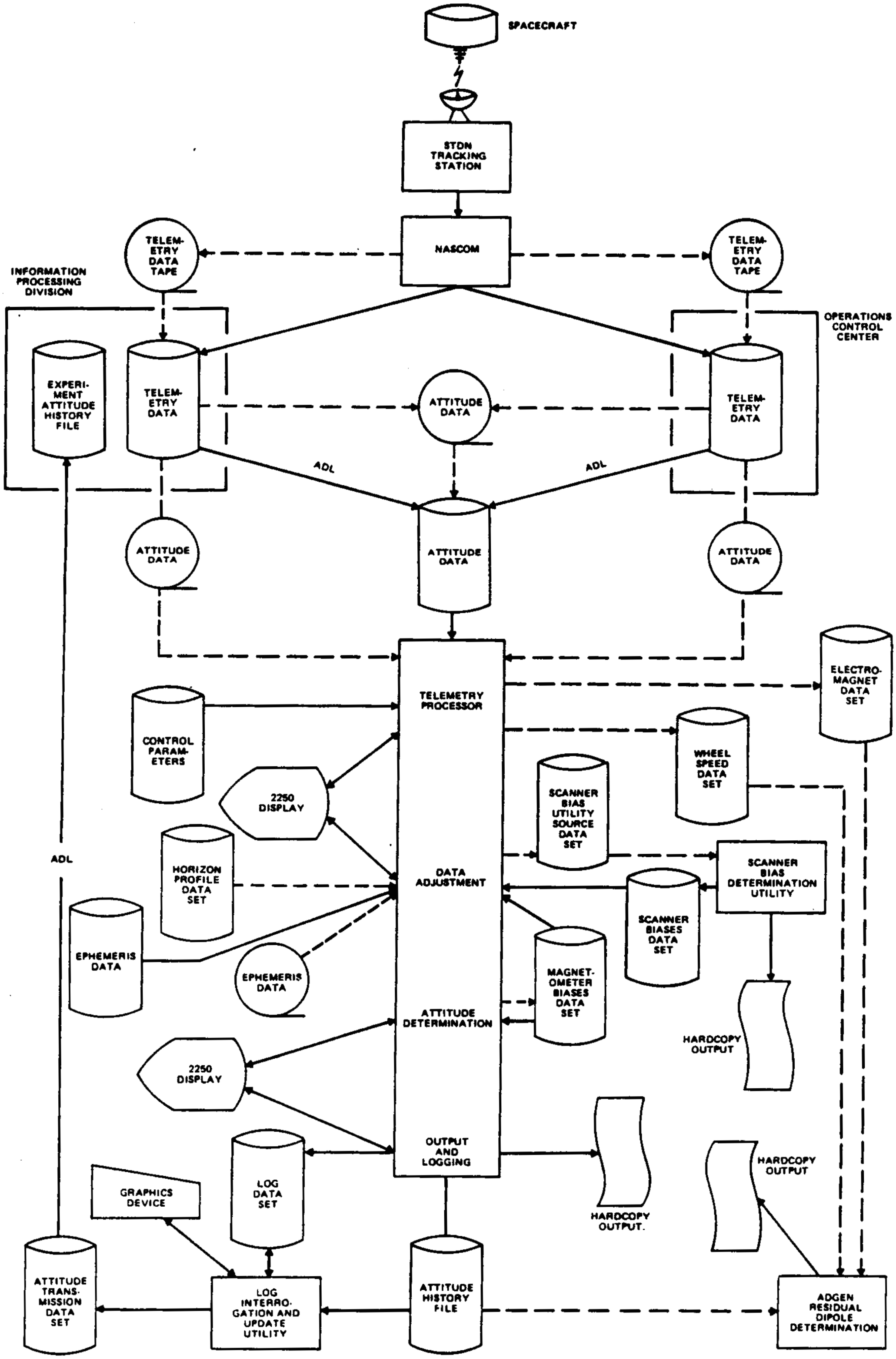


Figure 5-1. HCMM Attitude Determination and Control System Data Processing Flow Diagram

data routes are indicated by dashed lines in Figure 5-1). IPD then strips out relevant attitude telemetry data items, attaches header information (and time tags if required), and sends the resulting attitude telemetry data to the attitude determination computer via the Attitude Data Link (ADL) or backup magnetic tape. The telemetry processor then reads the telemetry data and NAMELIST control parameters and unpacks the attitude data. Some conversion to engineering units and quality checking of the data are performed. Bad quality data are flagged or deleted, and the remaining data are then available for validation and adjustment. Data are also placed into two permanently resident data sets for use by the ADGEN utility program. The first data set contains X-, Y-, and Z-axis electromagnetic dipoles and assoicated times; the second contains scan-wheel speed data and associated times. The first data adjustment functions are to convert the IR scanner duty cycle data to roll angle and to reject outlying sensor data using scalar and null attitude checks. Next, data is obtained at a user-specified time interval (normally every 10 seconds), and made available for attitude determination. To reject outlying data, spacecraft and solar ephemeris data are read from permanently resident ephemeris data sets or an ephemeris tape, and the magnetic field vector at each measurement time is computed. Ephemeris and magnetic field data are obtained at user-specified time intervals, together with the corresponding attached GMT time. On option, a permanently resident scanner bias utility source data set is written for use by the scanner bias determination utility program. This data set contains IR scanner pitch and roll angle data, the Sun vector in body and orbital coordinates, and associated times. The scanner bias determination program periodically produces a scanner biases data set which is required for data adjustment. This data set contains the current best estimates of the IR scanner pitch and roll biases, the uncertainties in these estimates, and the time when these estimates were determined.

The spacecraft attitude is computed at user-specified time intervals and optionally smoothed. Magnetometer biases may be computed for sunlit passes. The output and logging functions include summarizing the quantity and quality of the attitude solutions, cataloging the summary in a permanently resident log data set, and writing the solutions to an attitude history file data set. At appropriate intervals (normally twice per week), pertinent data from the attitude history file are loaded into the transmission data set and transmitted to IPD via data link or backup magnetic tape. These data are to be transmitted within 3 weeks from the time the ADCS receives definitive orbit tapes for the period covered.

The four major functions are discussed in detail in Sections 5.2 through 5.5. Section 5.6 describes the data sets required by the HCMM ADS.

5.2 TELEMETRY PROCESSING FUNCTIONS

The telemetry processing functions include reading the NAMELIST data sets, reading the telemetry data, extracting pertinent data from the telemetry record, converting certain items to engineering units, optionally checking the quality of the data, and generating data for further analysis.

5.2.1 Overview of Telemetry Processing Functions

Telemetry processing functions are shown in Figure 5-2 and are described in the following subsection.

- NAMELIST READING
- TELEMETRY DATA READING
- TELEMETRY DATA UNPACKING
- TELEMETRY DATA CONVERSION
- QUALITY CHECKING
- TIME CHECKING
- GENERATING DATA SAMPLES FOR FURTHER ANALYSIS

Figure 5-2. Telemetry Processing Functions

5.2.2 Input for Telemetry Processing Functions

Input for the telemetry processing functions consist of NAMELIST control parameters and telemetry data. Although all system NAMELIST data are read as part of the telemetry processing function, only two types are used for telemetry processing. The first type consists of control parameters which define the functions to be performed. These parameters determine, for example, what telemetry data to read, the level of debug hardcopy output to generate, what types of quality checking to perform, what type of time checking to perform, and whether to flag or delete bad data.

The parameters in this set minimally include the following:

- Biases added to converted values
- Flag values
- Which housekeeping status bits to process
- Debug and printout control parameters
- Telemetry input data set DDNAME
- Telemetry input data set device (disk/tape)
- Telemetry input data set format (OCC/IPD)
- Telemetry input data set access method (absolute record number or physical block and relative record number)
- Start and end record numbers
- Start and end block numbers
- Start and end volume numbers
- Start and end file numbers (tape)
- Read NAMELIST (yes/no)
- FORTRAN unit number for NAMELISTs
- Upper and lower limits of time interval to process
- Quality rejection criteria for sensor and control center quality flags
- Time quality rejection criteria (both GMT and spacecraft clock)
- Data correction techniques
- Disposition of bad quality data (flag/delete)
- Action to be taken for I/O errors and end-of-file (interrupt execution or continue reading)

The second NAMELIST used by the telemetry processor contains parameters required only if data are to be unpacked or converted in a manner other than nominal. Once the data format has been determined, these parameters will be defaulted to nominal values in a block data program, but this NAMELIST allows changes at execution time if required. See Sections 5.2.3.3 and 5.2.3.4 for more detail.

The telemetry input data can be read in four formats: (1) seven-track telemetry tape from the OCC, (2) nine-track telemetry tape from IPD, (3) telemetry data file on disk containing data from the IPD, and (4) telemetry data file on disk containing data from the OCC. Data in the last two formats are read using one of the two spacecraft identification numbers, one for OCC data and one for IPD data. The OCC data are read in 312-byte blocks; the IPD data, in 3492-byte blocks.

The format of these records is described in Section 5.6.2. One OCC record contains 4 minor frames, each of which contains one value of each type of sensor data. One IPD record contains 56 minor frames.

5.2.3 Description of Telemetry Processing Functions

5.2.3.1 Reading the NAMELISTs

All input parameters are read as part of the telemetry processing function. The individual NAMELISTs may reside in separate data sets or in a common data set in a prescribed order. Provision is made to detect incorrectly spelled or formatted parameters and to generate a display indicating on which NAMELIST display the incorrect parameter appears. This feature facilitates determining which parameter is in error and which other parameters did not acquire their assigned values. If an I/O error occurs or an end-of-file is encountered before reading all the sets of input parameters, appropriate graphic messages are generated.

5.2.3.2 Reading the Telemetry Data Set

If telemetry data are read based on their attached GMT rather than by record number, the telemetry data file is searched beginning with an input record number and continuing until a record is found with a GMT greater than the requested stop time. The record numbers of the first and last records containing GMTs within the requested time interval are returned to the calling program.

Data are read into a buffer area, which may be 312 (OCC data) or 3492 (IPD data) bytes long. Interface with the ADL routines includes appropriate error handling capabilities and recognition of the IPD Sentinel Record.

In the event of an I/O error, the record causing the error is ignored, an error message is diaplayed, and an attempt is made to read the next record. This process continues until a record is successfully read or a user-specified number of successive I/O errors are encountered. The latter event causes a graphic message to be displayed containing pertinent information such as the number of errors encountered and the last record number unsuccessfully read.

When loss of signal (LOS) occurs at a tracking station, no more data in that pass is processed since data from more than one station are not concatenated.

5.2.3.3 Unpacking the Telemetry Data

During the unpacking process, minor conversions are performed, such as converting the tracking station name from BCD to EBCDIC. The locations in the telemetry record and conversions to be performed are specified through NAMELIST parameters so they can be altered at execution time if desired. When necessary, the unpacked values are placed into temporary dynamically allocated arrays for storage until conversion takes place.

5.2.3.4 Telemetry Data Conversion

Conversion of telemetry data to engineering units, like the unpacking process, is specified through NAMELIST parameters, so the conversion process is subject to alteration at execution time. The conversion of magnetometer and scanwheel data is described in Section 4.2.1.1 and Sun data is described in Section 4.2.1.2. The GMT is converted from day of year and milliseconds of day to seconds since zero hours UT September 1, 1957.

5.2.3.5 Quality Checking

After conversion, the data are optionally checked for quality. Point-by-point checks include examination of the quality flag(s) attached by OCC or IPD and limit checking. Limit checking is provided for

- Coarse pitch
- Coarse roll
- Duty cycle I and II
- Scanwheel speed
- Electromagnet biases (x, y, and z)
- Roll bias

External flags are provided for

- Attached GMT
- OCC/IPD quality flag(s)

External flags are optionally provided for each data type subject to limit checks. No internal flagging is required.

In addition, any minor frame consisting entirely of zero fill data shall be deleted.

5.2.3.6 Scanwheel Speed Selection

There are two sources of scanwheel speed available in the MSOCC and IPD telemetry stream: a fine scanwheel speed, with a nominal range of 1750 to 2050 rpm, and a coarse scanwheel speed, with a nominal range of 600 to 2200 rpm. However, for purposes of calculating the spacecraft angular momentum and for writing the scanwheel speed data set, only one value of the scanwheel speeds may be used. The algorithm for selecting this one speed is

(1) select the fine scanwheel speed if it is less than a saturation value; otherwise, (2) select the coarse scanwheel speed. Note that both coarse and fine scanwheel speed data may be selected during a given pass.

5.2.3.7 Time Checking

Time checking can be performed at three levels of complexity at user option. The algorithms used for each level are discussed in Section 4.2.4 and consist of limit checks on the spacecraft clock and/or the attached GMT, correlation of the spacecraft clock with the GMT on a minor frame-by-frame basis, and correlation of successive values of the spacecraft clock or the GMT.

5.2.3.8 Building Output Data Segments

Data to be processed by the attitude determination functions consist of either all data processed, or only the unflagged data at the operator's option. They are displayed and, if desired, processing of a new data interval can be requested from these displays. Flagged values can be deleted automatically, based on external flag arrays containing more than one flag value, in addition to the standard GESS capabilities. The default flag value is a blank.

In definitive mode, the electromagnetic dipole data may be checked optionally to determine if wheel desaturation has occurred at any time in the processing interval. Wheel desaturation is indicated when either the first or third component of the electromagnetic dipole data differs significantly from the corresponding electromagnet bias value. The value of the wheel desaturation flag is set to indicate the presence or absence of wheel desaturation during the interval.

The expected volume of data per pass is approximately 600 values of each data item. The default data array size will hence be 600 if sufficient core is available. The data arrays require approximately 35K bytes of computer core in the nominal definitive mode, and approximately 65K bytes when all optional data types are included.

5.2.4 Output From the Telemetry Processing Functions

The output of the telemetry processing functions consists of the electromagnet data set and the wheel speed data set used by the ADGEN/HCMM utility program, converted telemetry data for the data adjustment functions, and hardcopy printout. The ADGEN/HCMM electromagnet data set and wheel speed data set supply electromagnet and scanwheel information for ADGEN to use in determining the residual magnetic dipole of the spacecraft. Both data sets are sequential. The first contains X-, Y-, and Z-axis electromagnet data and the associated time of the minor frame. The second data set contains scanwheel speed data and the associated time of the minor frame. Both data sets are updated at the minor frame rate; i.e., approximately every second. These control parameters are also available for display.

The data which are always generated include

- Attached GMT (milliseconds since zero hours UT September 1, 1957)
- X-, Y-, and Z-axis magnetometer reading (millioersteds)
- Sun sensor ID, N_a and N_b
- Scanwheel coarse pitch error data (degrees)
- Scanwheel fine pitch error data (degrees)
- Scanwheel duty cycle data (percent)
- Scanwheel fine roll error data (degrees)
- Scanwheel coarse roll error data (degrees)
- Flag arrays for the GMT and IPD or OCC quality flag(s)
- Scanwheel speed data (rpm)
- 24-byte header information obtained from the IPD telemetry header record

Data which can be generated on user option include

- Pitch command flag, indicating the direction of a 90-degree pitch maneuver
- Flag arrays for the magnetometer data, Sun sensor data, and scanwheel data
- Rate of change of the magnetic field (millioersteds per second)
- Wheel desaturation flag

Quality assurance parameters generated for the output and logging functions include

- Percent valid Sun data in the pass
- Percent valid magnetometer data in the pass
- Percent valid scanwheel data in the pass
- Percent valid attached GMT data in the pass

Data arrays which can optionally be unpacked, converted, and displayed during telemetry processing, but not used by the data adjustment functions, include

- X-, Y-, and Z-axis electromagnetic dipole data (pole-centimeters)
- Electromagnet biases (pole-centimeters)
- Commanded roll bias (degrees)
- Spacecraft housekeeping status bits, including a summary array

Printed output contains any of several levels of diagnostic printout for debugging purposes, controllable through NAMELIST parameters.

5.2.5 Telemetry Processing Displays

The telemetry processor optionally generates displays for diagnostic purposes. Additional displays are always generated in normal processing and are described in the following subsections. Pertinent output data are produced in hardcopy; GESS displays need not be printed.

5.2.5.1 NAMELIST Displays

The NAMELIST displays will contain all parameters input through NAMELIST, which are listed in Section 5.2.2. Only telemetry processor parameters are included in these displays, though all subsystem NAMELISTs are read at the same time.

5.2.5.2 Display of Input Telemetry Records

Two displays provide hexadecimal representations of (1) the OCC 312-byte telemetry record and (2) the IPD 3492-byte record. These displays are intended for initial diagnostic debugging purposes; each is a combined control parameter and tabular GESS display, as shown in Figures 5-3(a) and 5-3(b). The header variables are displayed as control parameters, and each minor frame of telemetry data is contained in one horizontal line of tabular hexadecimal characters, grouped in seven blocks of four bytes each. Only the 28 bytes of usable telemetry data are displayed; the 12 bytes of zero fill in each minor frame are omitted for display purposes. The OCC display contains 4 minor frames of telemetry data; the IPD display contains 56 minor frames. The column headings include the byte numbers of the data items in the first minor frame of the telemetry record in both cases.

5.2.5.3 Quick-Look Display

A display is provided during an intermediate processing stage of the telemetry processor for rapid examination of pertinent items in the telemetry data such as the attached GMT, the spacecraft clock, the minor frame number, and the OCC or IPD quality flag. This display is generated prior to converting any data other than those required for the display. An example of such a display is seen in Figure 5-3. This display includes the record number, the OCC quality flag, the telemetry format, the attached GMT, and the minor frame ID.

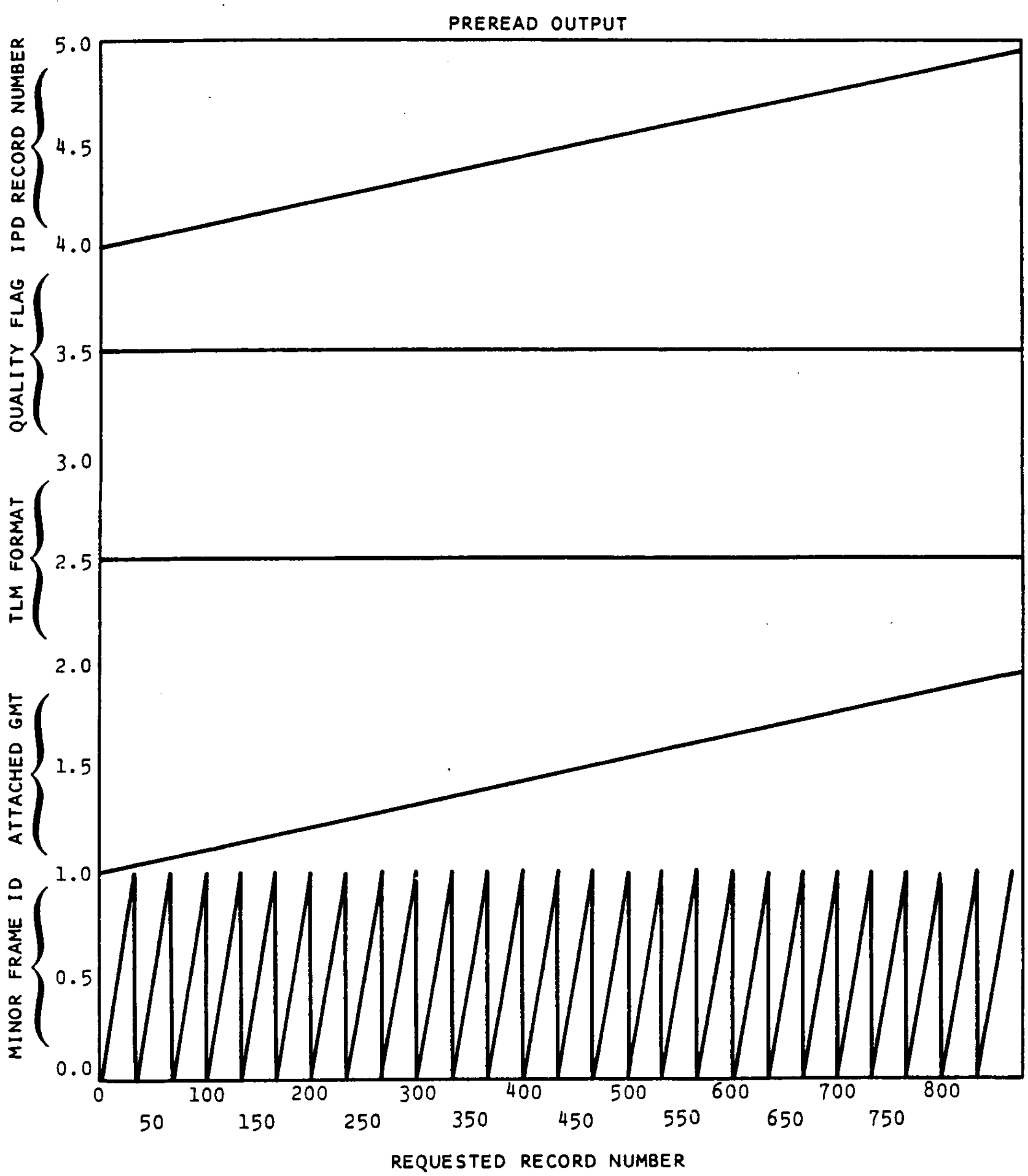


Figure 5-3. Sample Quick-Look Display

```
SPACECRAFT ID . . . . . . . . . . . . . . . . XX
DATA MODE . . . . . . . . . . . . . . . . . . XX
DATA TYPE . . . . . . . . . . . . . . . . . . XX
RECORD NUMBER . . . . . . . . . . . . . . . . XXXX
STATION NAME. . . . . . . . . . . . . . . . . XXXXXX
FILL  . . . . . . . . . . . . . . . . . . . . XXXX
PASS CONTACT NUMBER . . . . . . . . . . . . . XXXX
FILL  . . . . . . . . . . . . . . . . . . . . XXXXXXXX
AOS . . . . . . . . . . . . . . . . . . . . . XXXXXXXX
FILL  . . . . . . . . . . . . . . . . . . . . XXXXXXXX
```

BYTES 1-4	BYTES 5-8	BYTES 9-12	BYTES 13-16	BYTES 17-20	BYTES 21-24	BYTES 25-28
XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX
XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX
XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX
XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX	XXXXXXXX

Figure 5-3(a). Hexadecimal Display of OCC Telemetry Record

```
SPACECRAFT ID . . . . . . . . . . . . . . . . XX
DATA MODE . . . . . . . . . . . . . . . . . . XX
DATA TYPE . . . . . . . . . . . . . . . . . . XX
RECORD/BLOCK  . . . . . . . . . . . . . . . . XXXX
STATION NAME. . . . . . . . . . . . . . . . . XXXXXX
RECORD/TRANSMISSION . . . . . . . . . . . . . XXXX
PASS CONTACT NUMBER . . . . . . . . . . . . . XXXX
DATE (RE) PROCESSED . . . . . . . . . . . . . XXXXXXXX
BLOCK START TIME. . . . . . . . . . . . . . . XXXXXXXX
BLOCK STOP TIME . . . . . . . . . . . . . . . XXXXXXXX

 BYTES     BYTES     BYTES     BYTES     BYTES     BYTES     BYTES
  1-4       5-8       9-12     13-16     17-20     21-24     25-28

XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX
XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX
XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX
XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX  XXXXXXXX
   .         .         .         .         .         .         .
   .         .         .         .         .         .         .
   .         .         .         .         .         .         .
   .         .         .         .         .         .         .

                    (56 LINES AS ABOVE)
```

Figure 5-3(b). Hexadecimal Display of IPD Telemetry Record

The display is used to examine the quality of the data prior to extensive processing by the telemetry processor. If the data have been incorrectly time tagged or are of poor quality, determination can be made quickly and a retransmission requested from the Control Center without analyzing incorrect data.

5.2.5.4 Processing Loop Control Display

In interactive mode, a display is generated (in GESS GO status) during all processing loops in the telemetry processor. Its purpose is (1) to continually inform the user of the status of processing, and (2) to permit termination of a loop exceeding prescribed time limits. The second capability is used in non-interactive mode as well. The display is shown in Figure 5-4 and is available for use by other functions as well as telemetry processing.

In interactive mode, the display appears each time a processing loop is entered. The first parameter on the display tells what loop is being executed (each loop is assigned a number). The display is then updated at 30-second intervals. Two limiting parameters are displayed: the amount of Central Processing Unit (CPU) time allowed for the loop, and the number of times the loop is allowed to be executed. The CPU time used in the loop thus far, as well as the number of times the loop has been executed, are also displayed. Each time the loop is executed a dummy GESS CPOINT is called to allow intervention via program function key 30, which causes the GESS XSTOPS display to appear. The LOOPS display can then be placed on STOP status, and the user can terminate the loop and continue processing at a user-defined point in the code. When either limit is reached (allotted CPU time or number of loops), the display status is automatically changed to STOP and the parameter corresponding to DISCONTINUE LOOP is changed to YES. If the user wishes to discontinue the loop, he SKIPs from the display and program control exits the loop to a user-defined point in the code. If the user wishes to continue the loop, he changes YES to NO and SKIPs. The display automatically returns to GO status and continues the loop. The loop limits are changed to 9.0E10 seconds and 999999 loops, respectively; i.e., the loop is executed indefinitely.

```
LOOPS
        EXCESSIVE LOOP CONTROL DISPLAY

THE SYSTEM IS IN LOOP NUMBER............. 1

THIS DISPLAY WILL BE UPDATED
AT 30-SECOND INTERVALS.

NUMBER OF CPU MINUTES ALLOWED FOR LOOP... 0.1E 02

NUMBER OF ELAPSED CPU MINUTES IN LOOP.... 0.3E-83

NUMBER OF ALLOWED LOOPS.................. 10

NUMBER OF ELAPSED LOOPS..................  0

PRESS KEY 30 TO INTERRUPT LOOP;
SET STATUS OF THIS DISPLAY TO 'STOP',
THEN SKIP FROM XSTOPS.

DISCONTINUE LOOP................... (NO,YES) NO
```

Figure 5-4. Processing Loop Control Display

In noninteractive mode, the loop is exited when either of the loop limits is reached, and a hardcopy message denotes this occurrence.

5.2.5.5 Error Message Display

In the event that errors or abnormal conditions are detected during processing, an error message is displayed as shown in Figure 5-5. The display contains the following information:

- What error was detected (each error is assigned a number, and will be described in subsequent documentation in numerical order)
- What subroutine detected the error
- The probable cause of the error
- Suggested corrective action
- Likely result if corrective action is not taken

This message is available to all functions within the HCMM ADS.

5.2.5.6 Telemetry Processing Output Displays

The data generated for the data adjustment functions are displayed after all processing has been completed by the telemetry processing function. The output displays minimally contain

- The date of the beginning of the data
- The time of day of each data item
- All scanner pitch and roll angle data
- All scanner duty cycle data
- The Sun sensor N_a and N_b
- The Sun sensor ID

```
AEM ERROR MESSAGE NUMBER.............. 001

DETECTED BY SUBROUTINE................ TLMDRV

PROBABLE CAUSE OF ERROR............... TEST
                                       MESSAGE

SUGGESTED CORRECTIVE ACTION........... CONTINUE
                                       PROCESSING

RESULT OF ALT-END..................... CONTINUE
```

Figure 5-5. Error Message Display

- The magnetometer X-, Y-, and Z-axis magnetic field data
- External flag arrays for the GMT/spacecraft clock, magnetometers, and Sun sensors

On option, displays may also contain

- Electromagnetic dipole data
- Electromagnetic zero bias
- Scanwheel roll bias
- Rate of change of the Earth's magnetic field
- Scanwheel speed
- Plot of the difference between the fine roll voltage and the commanded roll voltage
- Plot of the difference between the coarse roll voltage and the commanded roll voltage
- Housekeeping information, minimally including
  - Sun sensor on/off
  - Acquisition mode on/off
  - Pitch maneuver on/off
  - Magnetometer on/off
  - Magnetometer in/out of control loop
  - Automatic reacquisition on/off

Both plot and tabular displays of all attitude-related data versus time and attached GMT time versus frame number are provided.

5.3 DATA ADJUSTMENT FUNCTIONS

The data adjustment functions include obtaining spacecraft, Sun, and magnetic field ephemeris, correcting IR scanner data for various sources of error, validating the data and rejecting outliers, and reducing the data volume.

5.3.1 Overview of the Data Adjustment Functions

Figure 5-6 describes the functions performed for data adjustment. The ephemeris generation function obtains inertial reference vectors from spacecraft and Sun ephemeris and a magnetic field model at user-specified intervals. The scanner data processing functions include correcting the scanner data for oblateness, CO_2 height, and eccentricity effects and selecting one value of the pitch and roll angle from the three sources available. Sun, Earth to spacecraft, and magnetic field vectors in body coordinates are then computed, based on Sun sensor, IR scanner, and magnetometer data, respectively. The validation checks consist of attitude independent and null attitude checks and smoothing. In interactive mode, the operator reviews all data and can, on option, flag any remaining data. The data volume is optionally reduced either by preaveraging the valid data at the attitude solution frequency, or by smoothing all valid data and evaluating the smoothing polynomial at the requested frequency. Parameters describing the quality of valid data are computed. IR scanner, Sun sensor, and ephemeris data are written to the scanner bias utility source data set if requested. The operator can optionally review and edit all output sensor data.

5.3.2 Input for the Data Adjustment Functions

Input for the data adjustment functions consists of NAMELIST parameters, sensor data obtained from the telemetry processor, and sensor biases from the magnetometer and scanner biases data sets.

- EPHEMERIS GENERATION
- SCANNER DATA PROCESSING
  - CORRECTION FOR VARIOUS SOURCES OF ERROR
  - SELECTION OF ONE PITCH AND ROLL ANGLE
- COMPUTATION OF BODY VECTORS
- DATA VALIDATION
  - ATTITUDE INDEPENDENT CHECKS
  - NULL ATTITUDE CHECKS
  - SMOOTHING FOR VALIDATION
- REDUCTION OF DATA VOLUME
  - PREAVERAGING
  - SMOOTHING
- OUTPUT

Figure 5-6. Data Adjustment Functions

The following are input from NAMELIST:

- Start and stop times of the processing interval
- Maximum difference between nominal orbit inclination and that obtained from spacecraft ephemeris
- Frequency of valid sensor data output
- Frequency of ephemeris update
- Direction of pitch maneuver (+90 degrees or -90 degrees)
- Observation time shift for each data type
- Maximum pitch deviation from null
- Maximum roll deviation from null
- Sun sensor mounting angles (three angles for each of the three sensors)
- Maximum error in null attitude check for Sun sensor data
- Magnetometer alignment matrix
- Maximum error for magnetic field magnitude check
- Maximum errors for the three dot product checks
- Maximum degree of smoothing for validation polynomial
- Maximum degree of smoothing for output polynomial
- Maximum deviation of data from smoothing polynomial for acceptance (standard deviations)

The following are the sensor data required for data adjustment and obtained from the telemetry processor at the minor frame rate:

- Sun sensor ID, N_a and N_b
- Three components of the magnetometer (millioersteds)

- Duty cycle data (percent)
- Fine roll error data (degrees)
- Coarse roll error data (degrees)

The following input is obtained from external data sets:

- Scanner pitch and roll biases
- Magnetometer biases
- Position and velocity vectors of the spacecraft in inertial coordinates

5.3.3 Description of Data Adjustment Functions

5.3.3.1 Ephemeris Generation Function

If Sun sensor data is available at any time during the pass, the inertial Sun unit vector is obtained at the start time of the pass and used for all processing in that pass. Also, the sensor to body transformation matrices for the three Sun sensors are computed from sensor mounting angles specified via NAMELIST. Spacecraft ephemeris is obtained from an ephemeris data set or tape in EPHEM format, or it is optionally generated internally. The inertial to orbital transformation matrix is computed from the spacecraft ephemeris using Equation (4-2). The inertial geomagnetic field vector is obtained from a magnetic field model.

The Earth to spacecraft and magnetic field vectors and the transformation matrix are evaluated at a user-specified frequency.

5.3.3.2 Scanner Data Processing Function

The scanner data processing function includes converting duty cycle data to equivalent pitch and roll angles; correcting all pitch and roll data for oblateness, orbit eccentricity, and horizon radiance variation effects; and selecting one value of the pitch and roll angle from the three sources (fine, coarse, and duty cycle-generated) available. The duty cycle conversions are detailed in Equations (4-9) through (4-19). The algorithm for correcting the pitch and roll data is given in Equations (4-20) through (4-43).

The selection of one pitch and roll angle and the addition of the constant pitch and roll biases are described in Equation (4-44).

5.3.3.3 Body Vector Computation Function

The body vector computation function adjusts the observation times and transforms corrected IR scanner, Sun sensor, and magnetometer data to the Earth to spacecraft, Sun, and magnetic field vectors into body coordinates using the best available estimate of sensor biases.

The Sun vector in sensor coordinates is computed from Sun sensor data (Equation (4-51)), transformed into body coordinates (Equation (4-53)), and stored in arrays.

The magnetic field vector in body coordinates is computed from the magnetometer data (Equation (4-47)), and magnetometer biases, which are obtained from the magnetometer biases data set. The unbiased pitch and roll angles determine the Earth to spacecraft vector in body coordinates (Equation 4-46)).

5.3.3.4 Validation Function

This function provides for the validation and rejection of sensor data. The validation checks are divided into three categories. The first consists of those which do not require an estimate of the spacecraft attitude. These checks are made during all phases of the mission. The second consists of null attitude checks, which assume that the attitude is near nominal (pitch, roll, and yaw equal zero). These checks are made only during orbit adjust and mission modes. Finally, during all phases of the mission data can be smoothed for validation.

Figure 5-7 illustrates the functions performed during validation, indicates the order in which these functions are executed, and references the equations in Section 4 pertinent to each validation check.

Simple limit checks can be performed without knowledge of an a priori attitude. The Sun sensor data is flagged if the Sun sensor identification is invalid. The

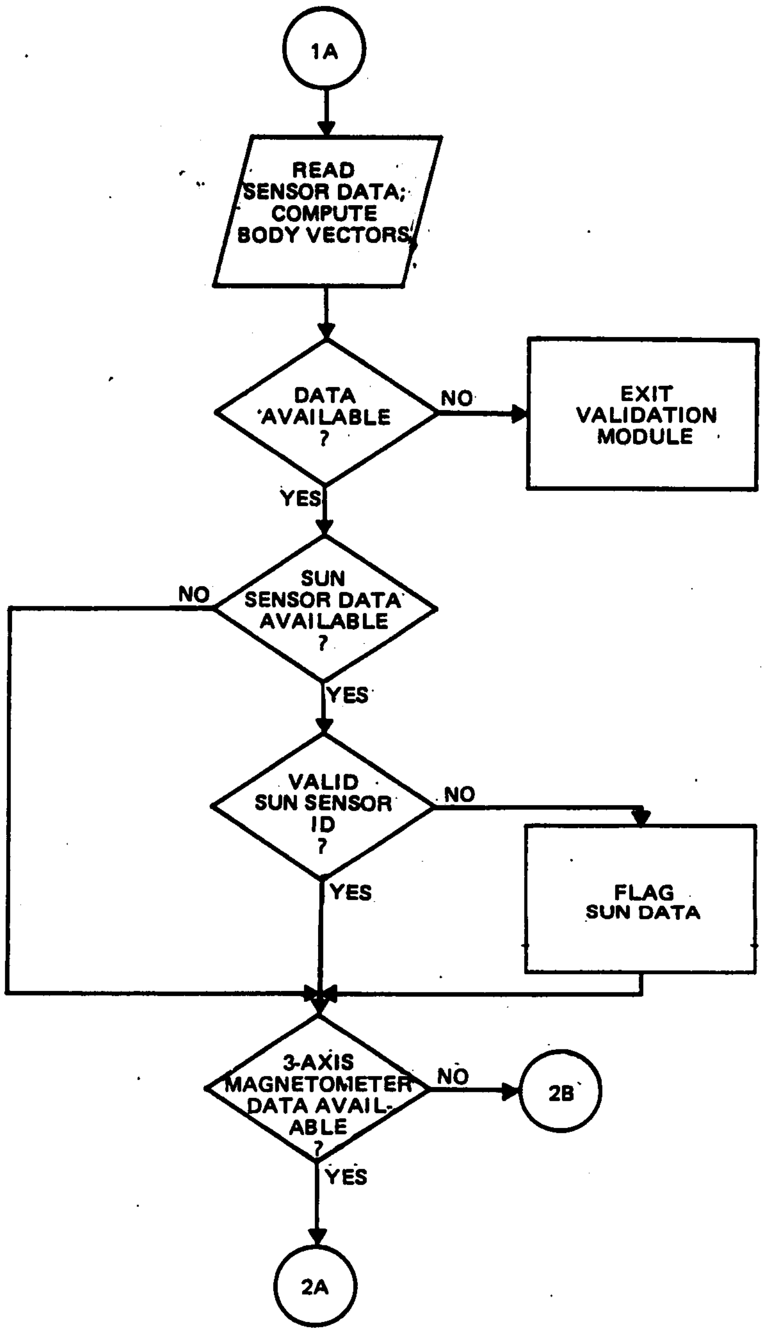


Figure 5-7. Validation Function Flow Diagram (1 of 5)

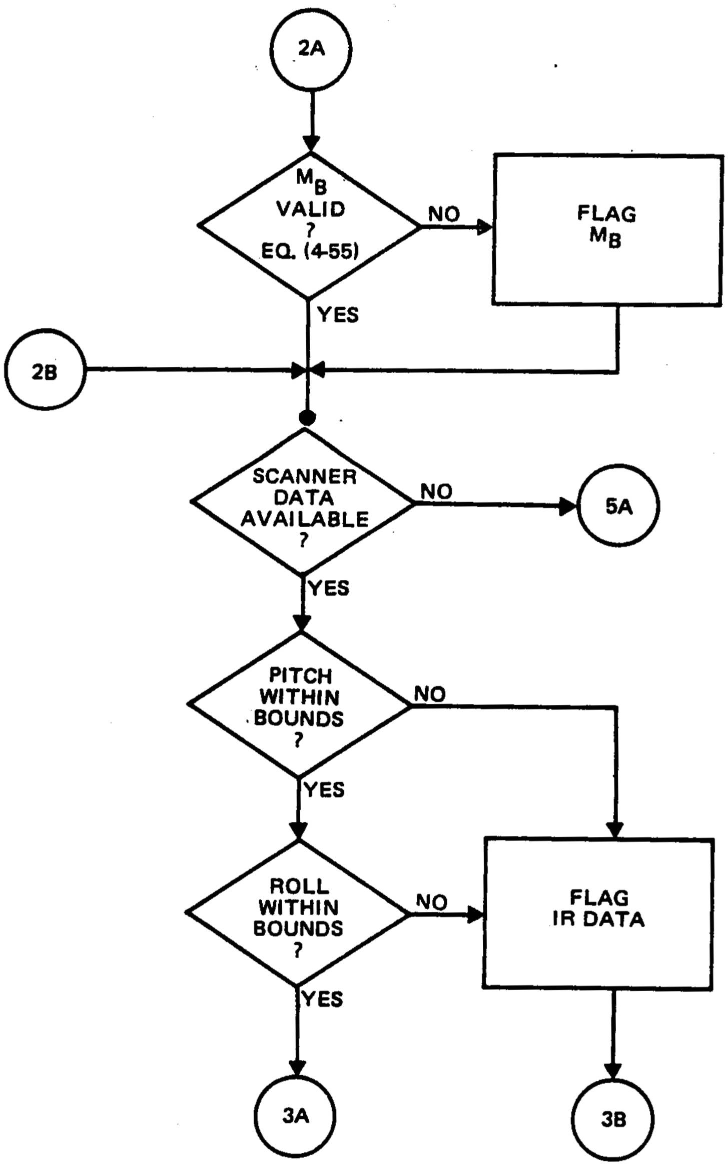


Figure 5-7. Validation Function Flow Diagram (2 of 5)

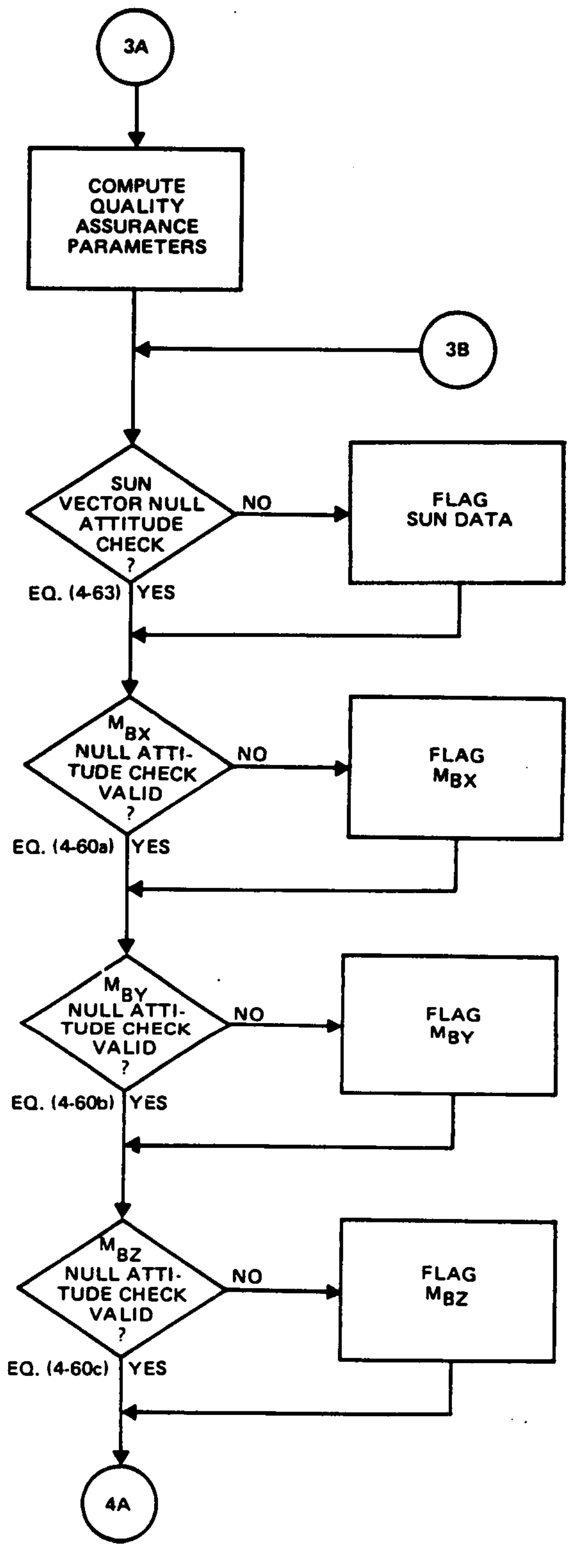


Figure 5-7. Validation Function Flow Diagram (3 of 5)

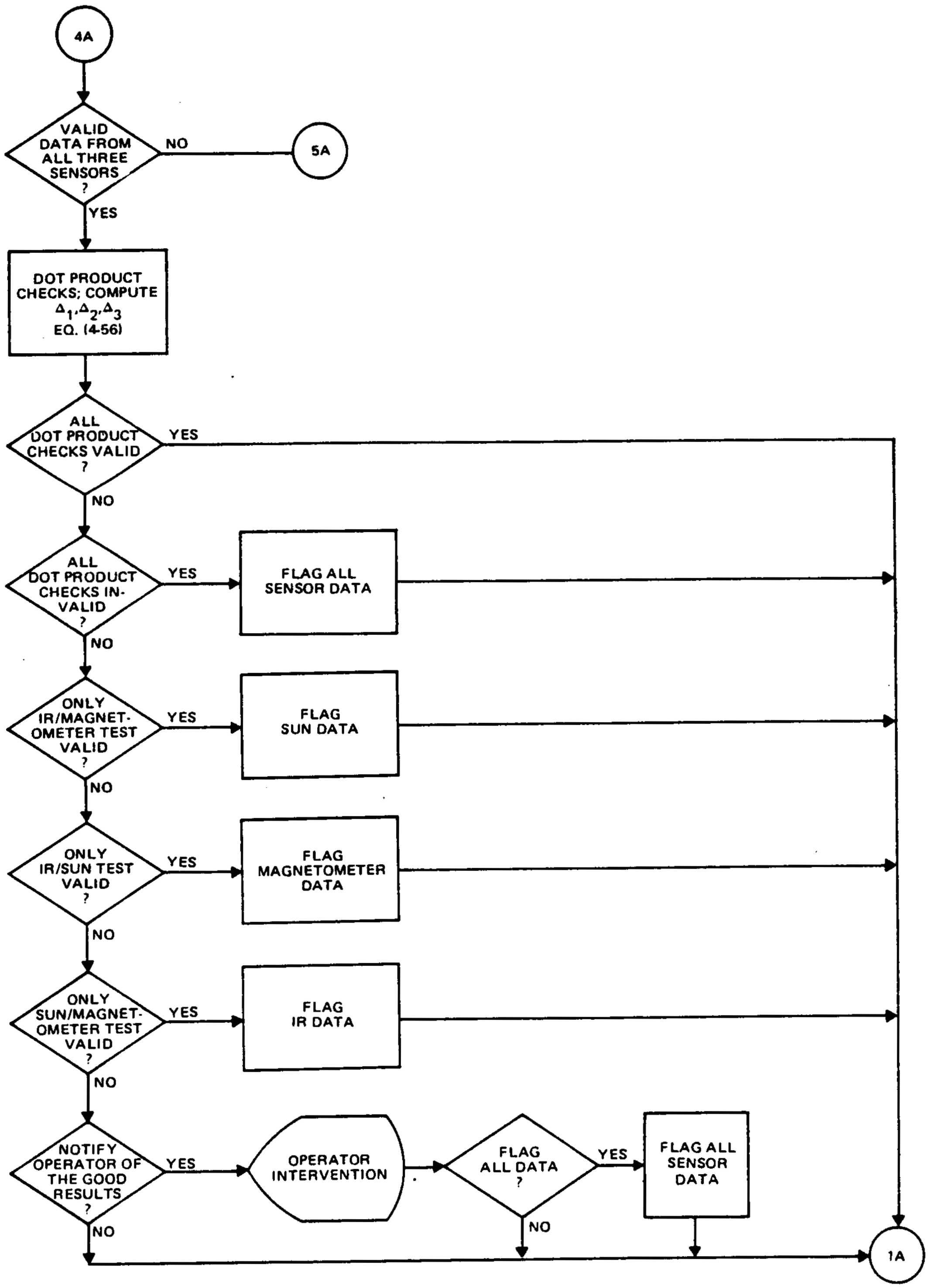


Figure 5-7. Validation Function Flow Diagram (4 of 5)

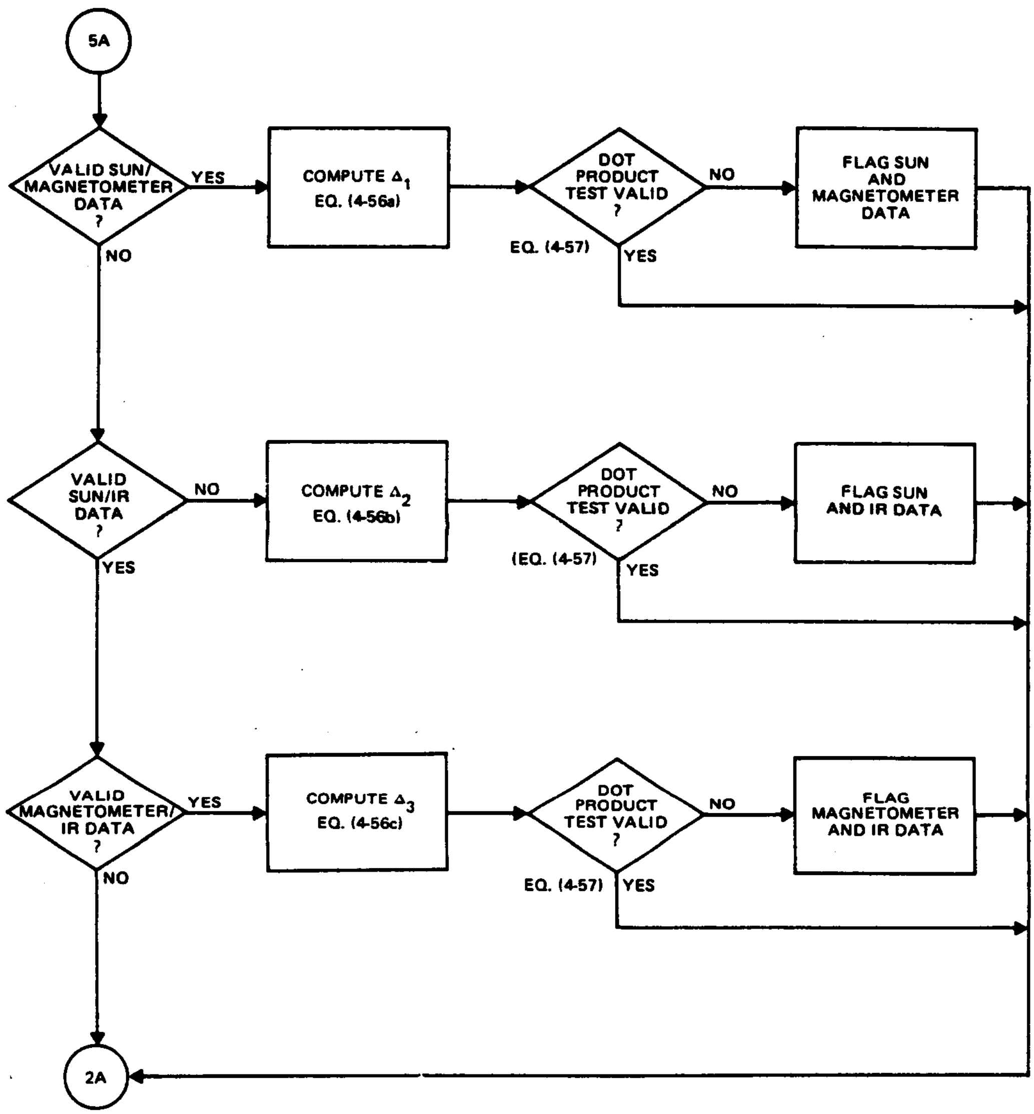


Figure 5-7. Validation Function Flow Diagram (5 of 5)

magnitudes of the model and observed magnetic field vector are compared. If there is a difference larger than a user-specified tolerance, the magnitude is flagged externally, but the output from each magnetometer axis is not internally flagged, pending further checks.

In on-orbit mode, the next step in the validation process is the null attitude check. This attitude dependent check assumes that the onboard control system will maintain the spacecraft to within about 3 degrees of the null attitude. In this case, the body and orbital coordinate systems coincide approximately, and a large discrepancy between the orbital and body vectors is attributed to an error in the corresponding sensor data.

The scanner pitch and roll angles are compared to user-specified tolerances which reflect both the maximum expected deviations from null attitude experienced by the spacecraft and the uncertainties in the pitch and roll biases. If either the pitch or roll angle is greater than its corresponding tolerance, both data are flagged.

Sun sensor data is checked by computing the angle between the orbital and body Sun unit vectors. If this angle is larger than a specified tolerance, the Sun data is flagged. The Sun sensor ON/OFF flag appears in bit 4 in telemetry word 77. If the value of this flag is 0 (off), the Sun sensor data (ID, NA, and NB) for the minor frame should, on option, be flagged. If the value is 1 (on), it should not be flagged.

Unlike the Sun sensor data, each component of the magnetic field vector is checked separately, because attitude solutions can be computed from degraded magnetometer data. If the difference between corresponding components of the orbital and body magnetic field vectors is greater than a user-specified tolerance, that component is internally flagged. This tolerance reflects not only deviations of the attitude from null, but also uncertainties in the magnetometer biases, magnetometer misalignments, and errors in the magnetic field model which provided the inertial magnetic field vector.

Dot product checks are made for all combinations of two valid data types. In acquisition mode when only Sun sensor and magnetometer data are available, and in mission mode when only two valid data types are available, both types of data are flagged if this test fails. This is done because there are no validation tests to indicate which of the two data types is the cause of the failure. However, when all three data types are available, it is possible to determine from the dot product checks which, if any, data are invalid in the following way. If the data passes all the scalar checks, no data is flagged. If the data fails all three checks, at most one data type is good, but all are flagged because one data type alone cannot be used by the attitude determination subsystem. If the data fails two of the three dot product checks, the data type which is common to the two failed combinations is flagged. The remaining result (one of the three checks fails) is logically inconsistent and is probably due to unrealistic error criteria. In this case, no data is flagged. In interactive mode, the operator is notified and given the option to reexecute the validation function with different error criteria.

Sensor data is smoothed for validation in mission mode and optionally during attitude acquisition. This function is performed at the end of the pass, after all available data have been checked by previously defined techniques. Smoothing is done for valid IR scanner pitch and roll, the components of $\bar{M}_B$, N_a , and N_b . Each of these data types is fitted to a sum of orthogonal polynomials with a user-specified number of terms. Data points which deviate from the polynomial by more than a user-specified number of standard deviations are flagged. This process is repeated for the remaining valid data until no data are rejected.

In interactive mode, all valid data are displayed, and the operator may delete any data or reexecute the validation function using different rejection criteria as desired.

5.3.3.5 Data Volume Reduction Function

The data volume may optionally be reduced from one sample per minor frame (≈1 second) to one sample at the attitude solution period (≈10 seconds). This is accomplished either by preaveraging or by smoothing. Preaveraging, which is discussed in Section 4.2.5, requires subdividing the data into time intervals equal to the attitude solution period and calculating the average value of each data type with the intervals. In smoothing, a data type is fit to a sum of orthogonal polynomial(s) with a user-specified maximum number of terms over the processing interval. The resulting function is then evaluated at the attitude solution period.

The data to be processed in this fashion are the IR scanner pitch and roll angles which have been corrected for all effects, the three components of the magnetic field vector in body coordinates, and the Sun sensor N_a and N_b.

Values of N_a and N_b are used to construct the Sun vector in body coordinates. However, if the ID changes over an interval, an average N_a or N_b will be meaningless because it is a sensor dependent number. In this case, Sun data is processed by preaveraging or smoothing the components of the Sun vector directly. The validation function may be reexecuted using a different data reduction technique.

5.3.3.6 Output Function

The output function provides for the computation of all quality assurance parameters. For each data type, rms deviations from expected values in the null attitude, magnitude, and dot product checks are computed. The length of the largest gap in each data type is also determined.

In addition, scanner and Sun sensor data are optionally written to the scanner bias utility source data set at the attitude solution period. In interactive mode, all output data are optionally displayed.

5.3.4 Output From Data Adjustment Functions

Output from the data adjustment functions consists of validated attitude sensor and ephemeris data and quality assurance parameters, hardcopy printout, and optionally, attitude sensor and ephemeris information to the scanner bias utility source data set.

In obtaining the values of the magnetometer and constant scanner biases for data processing, an option shall be provided either to read these values from the magnetometer biases and scanner biases data sets or to use the default values provided via NAMELIST.

The validated attitude sensor and ephemeris data which are generated at the attitude solution frequency consist of

- Time of data sample (seconds since zero hours UT September 1, 1957)
- IR scanner pitch, with constant pitch bias applied (degrees)
- Type of pitch data selected (fine, coarse, duty cycle-generated)
- IR scanner roll, with constant roll bias applied (degrees)
- Type of roll data selected (fine, coarse, duty cycle-generated)
- Sun sensor ID, N_a, and N_b
- Sun unit vector in body coordinates
- Magnetic field vector in body coordinates, without constant biases applied (millioersteds)
- Spacecraft velocity vector (kilometers per second)
- Earth to spacecraft vector (kilometers)
- Inertial magnetic field vector (millioersteds)
- Inertial Sun unit vector
- Scanwheel speed (rpm)

The quality assurance parameters, which are calculated only in definitive mode, include

- Percentage of good scanner data after data adjustment checks (scalar checks, dot product checks, null attitude checks)
- Percentage of good Sun data after data adjustment checks
- Percentage of good magnetometer data after data adjustment checks
- Largest gap in scanner data (seconds)
- Largest gap in Sun data (seconds)
- Largest gap in magnetometer data (seconds)
- Quality of valid scanner data (rms difference from null attitude)
- Quality of valid Sun data (rms difference from null attitude)
- Quality of valid magnetometer data (rms difference from null attitude)
- Quality of valid magnetometer data (rms difference from magnetic field magnitude)
- Quality of valid Sun/magnetometer data (rms difference of dot product check)
- Quality of valid Sun/IR data (rms difference of dot product check)
- Quality of valid magnetometer/IR data (rms difference of dot product check)

The following data are optionally written to the scanner bias utility source data set at the attitude solution frequency:

- Time of data (seconds since zero hours UT September 1, 1957)
- Sun unit vector in body coordinates
- Third component of the Sun unit vector in orbital coordinates

- IR scanner fine pitch and roll angles (degrees)
- IR scanner coarse pitch and roll angles (degrees)
- IR scanner duty cycle-generated pitch and roll angles (degrees)

Printed output, in both near-real-time and definitive modes, contains any of several levels of diagnostic printout for debugging purposes, controllable through NAMELIST parameters.

5.3.5 Data Adjustment Displays

Displays are generated during the data adjustment functions to allow the operator to monitor all input and output. Other displays permit the operator to monitor and control the execution of various functions. During interactive mode, the following tabular displays and data plots are provided.

5.3.5.1 NAMELIST Display

The NAMELIST displays will contain all input NAMELIST parameters, which are listed in Section 5.3.2.

5.3.5.2 Data Adjustment Input Displays

Displays of all data required for data adjustment are available on option. These displays include

- Sensor data passed from the telemetry processor
- Quality assurance parameters generated by the telemetry processor
- Current values of the scanner pitch and roll and the magnetometer biases
- Contents of the scanner biases and magnetometer biases data sets

Both tabular displays and plots of the data obtained from the telemetry processor are provided.

5.3.5.3 Data Adjustment Output Displays

Displays of data arrays required for attitude determination and parameters which indicate the effectiveness of the data validation and data reduction function are provided. These displays include

- All validated sensor data
- Quality assurance parameters
- All data written to the scanner bias utility source data set
- Tables of all data rejected during magnitude, dot product, or null attitude checks, with a parameter indicating the failed checks
- A plot of the raw and calibrated sensor data and smoothing polynomial versus time, with rejected data flagged on the plot
- A plot comparing the components of magnetic field rate measured by the control system with the components of the magnetic field rate calculated from magnetometer data
- Plot of computed and observed (based on both raw and calibrated data) magnetic field magnitude versus time
- Plot of Sun azimuth and elevation in body coordinates
- Plot of the field-of-view of the Sun sensors

The Sun sensor field of view plot is shown in Figure 5-7(a).

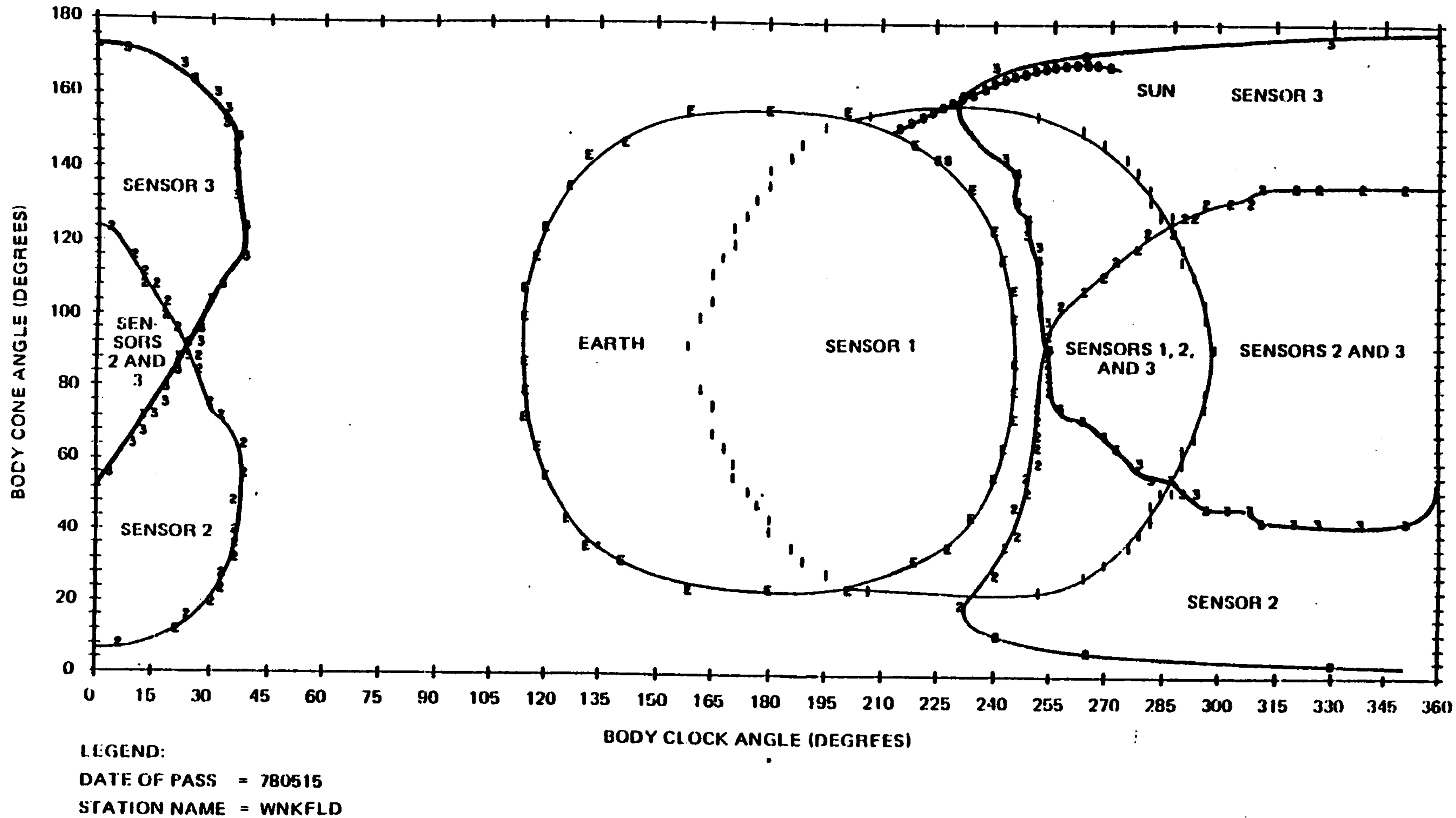


Figure 5-7(a). Sun Sensor Field of View

5.4 ATTITUDE DETERMINATION FUNCTIONS

The attitude determination functions include determining the magnetometer residual biases, using validated sensor data to calculate three-axis attitude and attitude rate solutions, determining the uncertainties in the attitude solutions, smoothing the solutions for output purposes, and calculating output parameters.

5.4.1 Overview of the Attitude Determination Functions

Figure 5-8 describes the functions performed for attitude determination. The initializing function includes optionally displaying the NAMELIST parameters and thus permitting alterations to the NAMELIST in interactive mode. The method of attitude determination (discrete or differential corrector) is selected via NAMELIST or, optionally, is based on the amount of available sensor data.

The magnetometer bias determination function calculates the residual biases on each magnetometer for sunlit passes in definitive mode, writes the results to the magnetometer biases data set, and displays the biases.

The discrete attitude determination function includes determining a three-axis attitude solution on a point-by-point basis using all possible combinations of infrared scanner, Sun sensor, and magnetometer data. In acquisition mode, this function includes computing and displaying pitch, roll, and yaw rates and the angle between the spacecraft Y-axis and the negative orbit normal. Also included during definitive operations is calculating the uncertainty for each attitude solution based on the estimated errors in the measurements and the reference vector geometry.

The differential corrector function is used only in definitive mode and only when the user has requested its use, or when it has been determined that there is insufficient sensor data for discrete attitude determination. This function assumes a simple polynomial model for the time dependence of the attitude and uses sensor data to estimate both the coefficients of this polynomial and, optionally, residual magnetometer biases.

The smoothing function is used both in acquisition and definitive modes. If the discrete attitude determination method was used, all the attitude solutions for a given time are combined to obtain a weighted average attitude and the associated uncertainty. The resulting attitude solutions are smoothed by fitting the pitch, roll, and yaw angles to polynomials of a user-specified maximum degree and the quality assurance parameters are calculated.

5.4.2 Input for the Attitude Determination Functions

Input for the attitude determination functions consists of validated sensor data from the data adjustment functions, control parameters via NAMELIST, and the magnetometer biases data set.

The attitude data obtained from the data adjustment and telemetry processing functions contain validated sensor and control system output and ephemeris data at a user-specified frequency. This data consists of

- Time of data sample
- IR scanner pitch, with constant pitch bias applied (degrees)
- Type of pitch data selected (fine, coarse, duty cycle-generated)
- IR scanner roll, with constant roll bias applied (degrees)
- Type of roll data selected (fine, coarse, duty cycle-generated)
- Sun sensor N_a and N_b
- Sun sensor ID
- Sun unit vector in body coordinates
- Magnetic field vector in body coordinates, without constant biases applied (millioersteds)
- Scanwheel speed (rpm)

- INITIALIZATION
- MAGNETOMETER BIAS DETERMINATION
- DISCRETE ATTITUDE DETERMINATION
  - ATTITUDE COMPUTATION
  - ATTITUDE RATE COMPUTATION
  - ERROR STATISTICS COMPUTATION
- DIFFERENTIAL CORRECTOR
  - ESTIMATION OF ATTITUDE PROPAGATION COEFFICIENTS
  - ESTIMATION OF ERRORS
- SMOOTH ATTITUDE SOLUTIONS
- OUTPUT

Figure 5-8. Attitude Determination Functions

- Spacecraft velocity vector (kilometers per second)
- Earth to spacecraft vector (kilometers)
- Inertial magnetic field vector (millioersteds)
- Inertial Sun unit vector

Optional data include the X-, Y-, and Z-axis magnetometer rates and a wheel desaturation flag.

The input from the magnetometer bias data set consists of X-, Y-, and Z-axis magnetometer biases and associated uncertainties.

The NAMELIST parameters include the following:

- Method of attitude determination (discrete/differential corrector)
- Magnetometer bias determination (yes/no)
- Minimum angle between the Sun vector and spacecraft yaw axis for magnetometer bias determination
- Minimum number of Sun sensor and magnetometer data samples for magnetometer bias determination
- Criteria for valid magnetometer bias solutions
- Largest gap in attitude sensor data permitted for discrete solutions
- Criteria for flagging as degraded components of the magnetometer data
- Solve-for parameters of the differential corrector state vector
- Measurement weights for the differential corrector
- Maximum number of iterations for the differential corrector
- Maximum uncertainty for valid differential corrector solutions

- Attitude solution smoothing (yes/no)
- Maximum degree of polynomials used for smoothing
- Maximum value of goodness-of-fit parameter associated with smoothing polynomial for valid solutions
- Frequency of output for attitude solutions
- Default weights for discrete attitude solutions
- Spacecraft moment of inertia tensor
- Wheel moment of inertia
- FORTRAN unit number for magnetometer biases data set
- Debug and printout control parameters

5.4.3 Description of Attitude Determination Functions

5.4.3.1 Attitude Determination Initializing Function

The attitude determination initializing functions include displaying the subsystem NAMELIST in interactive mode and allowing the operator to alter NAMELIST parameters. A hardcopy printout of the NAMELIST is produced. In acquisition mode, the presence of Sun sensor data is checked for. If there is none, attitude and attitude rate determination is not possible, and the remainder of the attitude determination functions are not performed. If Sun sensor data are available, discrete attitude solutions are computed. In definitive mode, NAMELIST parameters determine if magnetometer bias determination is to be performed. If it is to be done, it is verified that adequate Sun sensor data which satisfies a user-specified geometry condition is available. If discrete attitude determination has been requested, this function, on option, first verifies that the largest gap in attitude sensor data is smaller than a user-specified time before permitting discrete attitude determination. If the data gap is larger, the differential corrector method is used to compute attitudes. The method of attitude determination and a flag indicating whether magnetometer bias determination is to be performed is displayed in interactive mode, and a hardcopy printout of this information is generated.

5.4.3.2 Magnetometer Bias Determination Function

Magnetometer bias determination is performed in definitive mode whenever the user has requested this option and when the initializing function has verified that there is sufficient Sun sensor and magnetometer data. The algorithm, which uses IR scanner pitch and roll data and Sun sensor data to obtain estimates of the residual biases on the X-, Y-, and Z-axis magnetometers and the standard deviations for these estimates, is given in Section 4.4.2. After the values for the magnetometer biases and the associated standard deviations have been computed, the magnitudes of the standard deviations are compared to a user-specified value. If all standard deviations are larger than this value, the magnetometer biases data set is not updated. If any standard deviation is smaller than this value, the new biases are compared to the most recently determined values residing in the magnetometer biases data set. If any new bias differs from the previous bias by more than a user-specified tolerance, the data set is not updated. However, if both data quality checks are satisfied, the values of the new biases and associated standard deviations are optionally written onto the magnetometer biases data set. Hardcopy printout of the calculated biases and standard deviations, the user-specified tolerances which were used, and the final contents of the magnetometer biases data set is provided. In interactive mode, this same information is displayed, and the capability to redetermine the biases with different user-supplied tolerances is provided.

5.4.3.3 Discrete Attitude Determination Function

Discrete attitude solutions are computed in acquisition mode and during definitive operations when directed by the initializing function. Figure 5-9 describes the computational flow for attitude determination. This figure indicates the

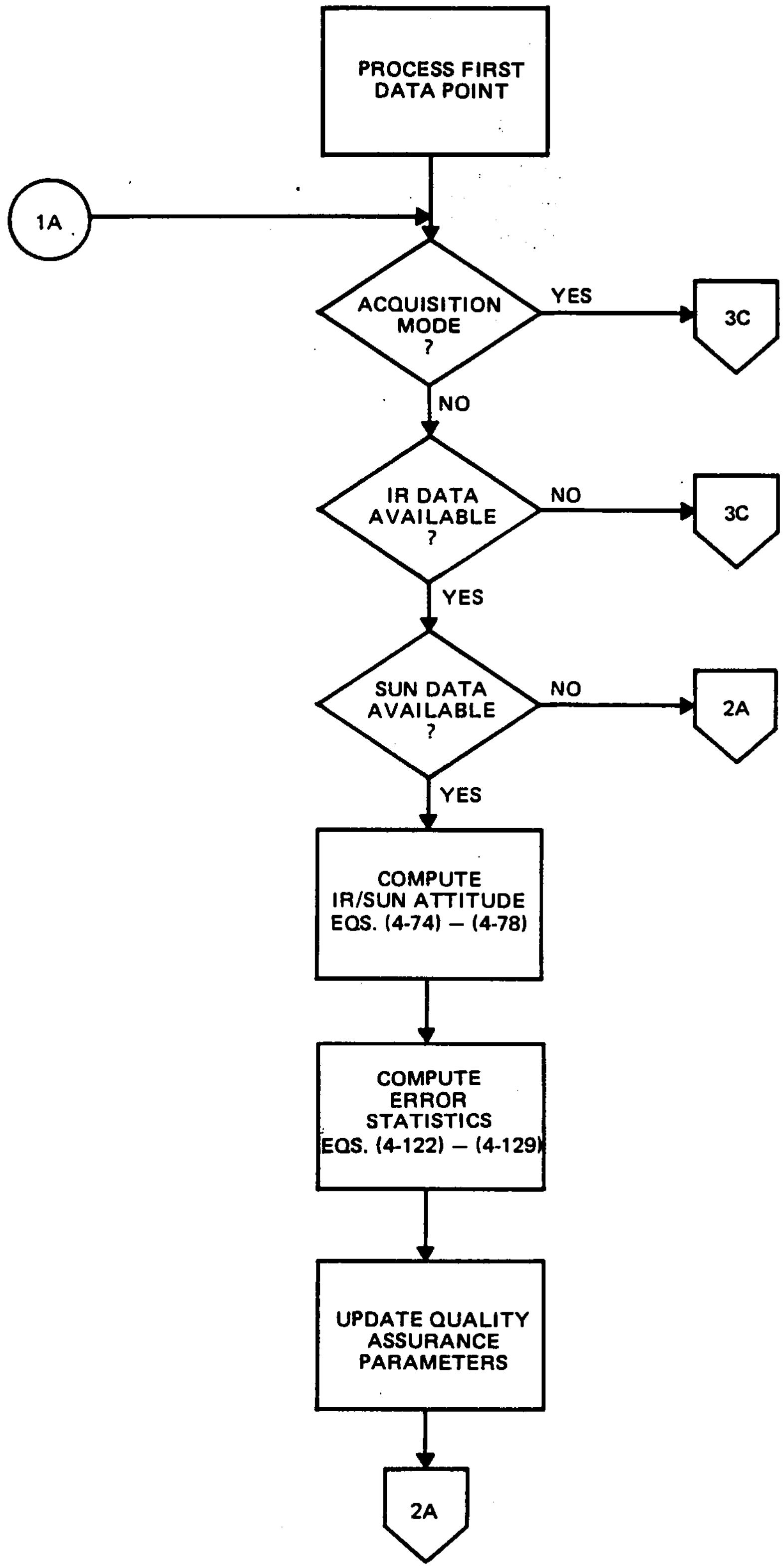


Figure 5-9. Computational Flow of Discrete Attitude Determination Function (1 of 5)

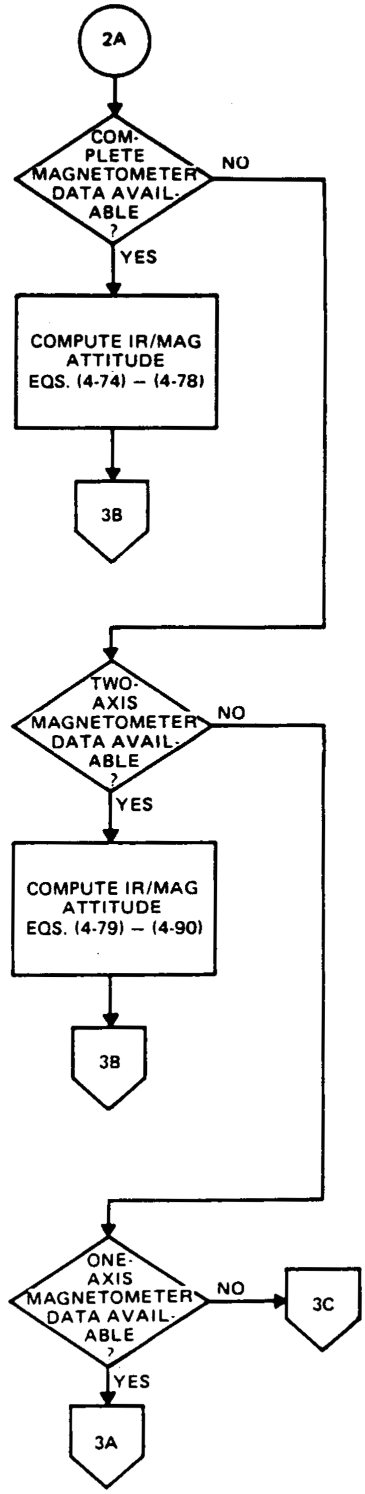


Figure 5-9. Computational Flow of Discrete Attitude Determination Function (2 of 5)

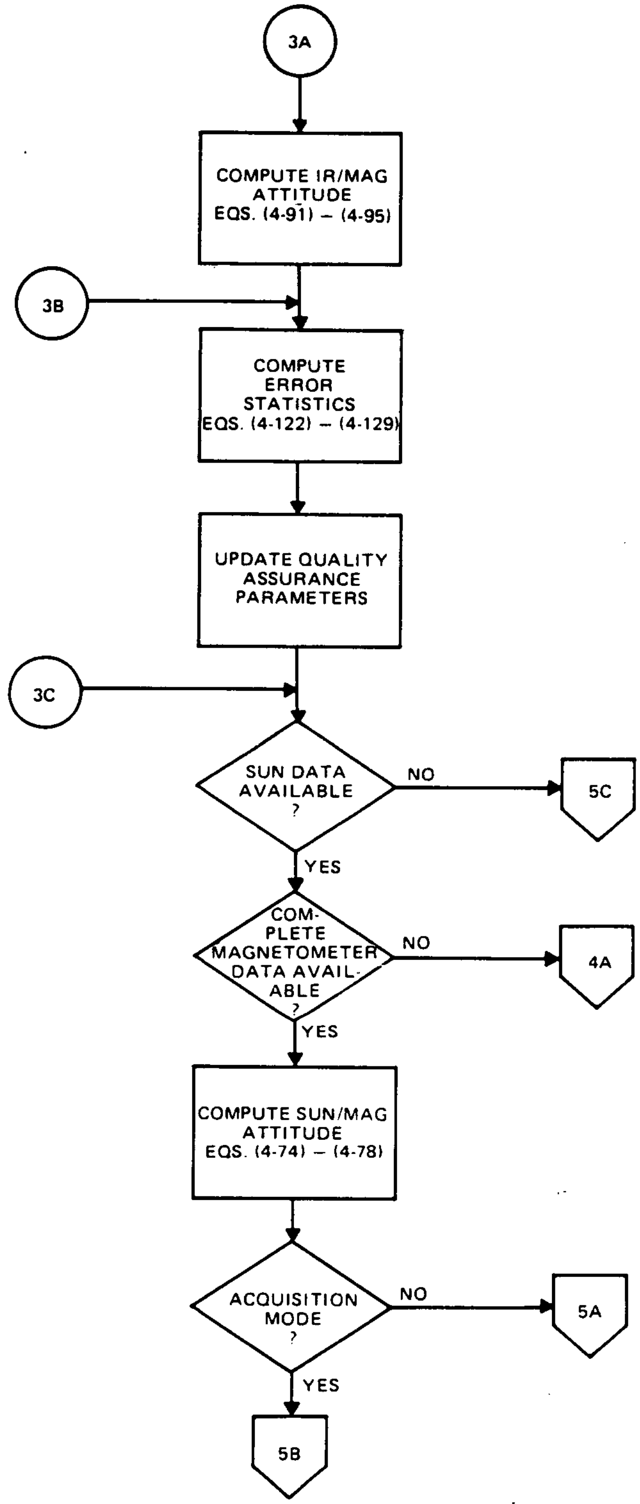


Figure 5-9. Computational Flow of Discrete Attitude Determination Function (3 of 5)

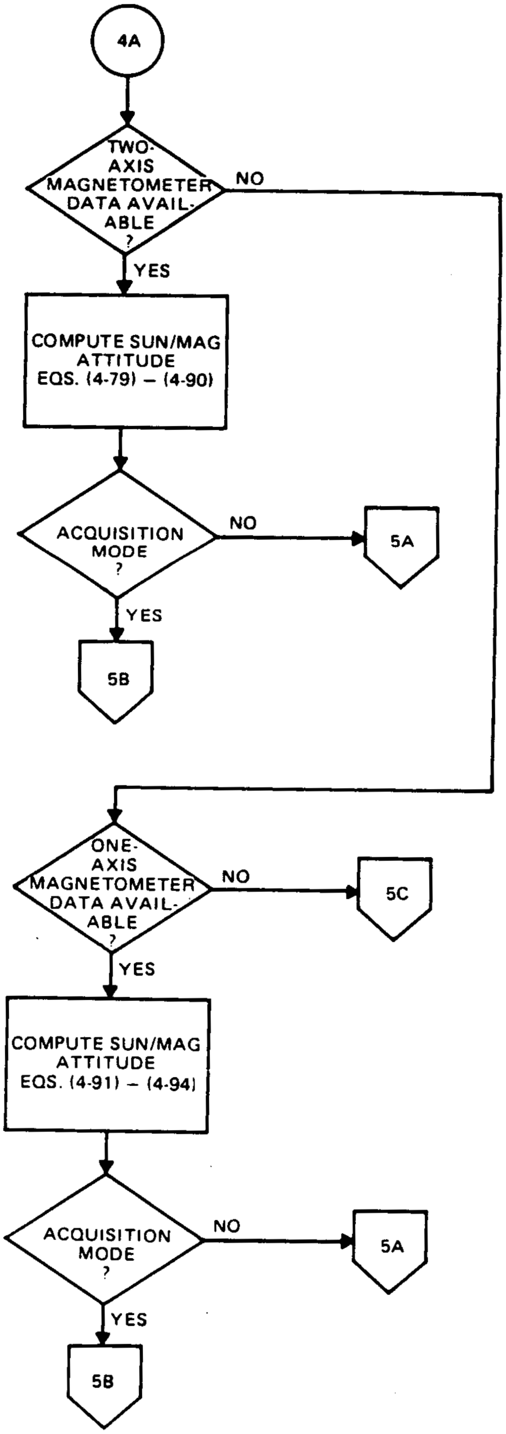


Figure 5-9. Computational Flow of Discrete Attitude Determination Function (4 of 5)

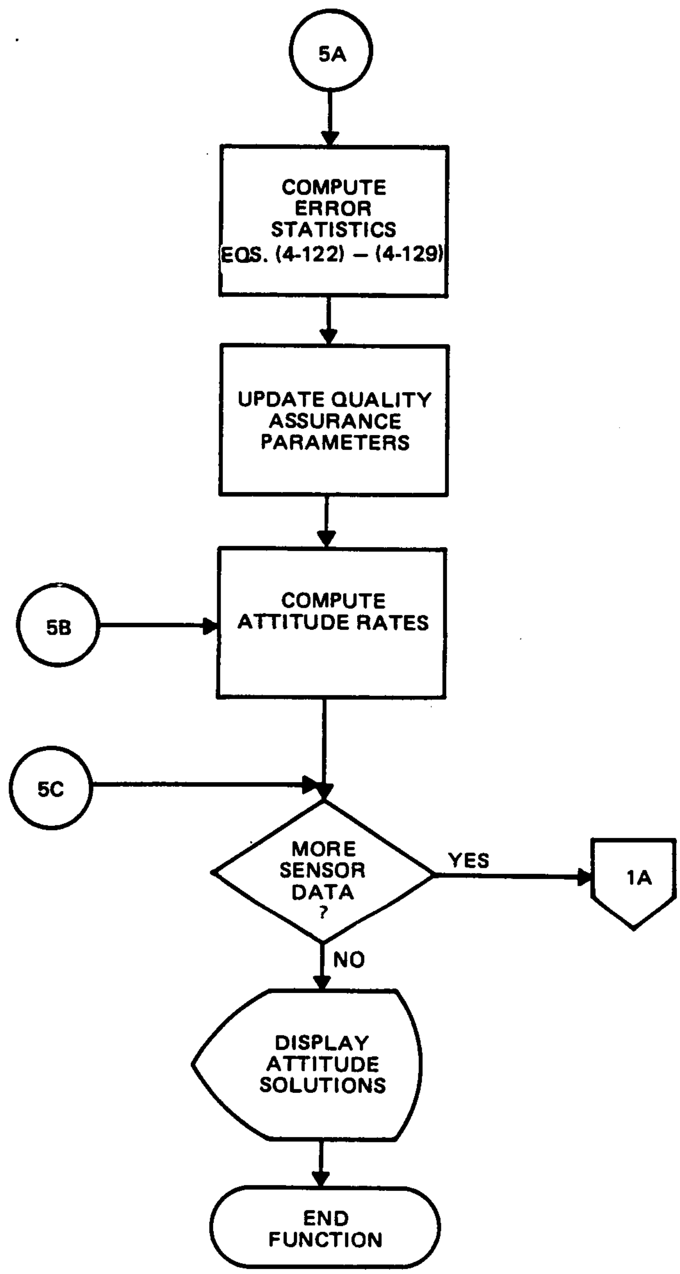


Figure 5-9. Computational Flow of Discrete Attitude Determination Function (5 of 5)

combinations of sensor data used to obtain attitude and attitude rate information, and which equations in Section 4 are pertinent to the computation.

In acquisition mode, it is possible to determine the attitude from complete Sun sensor data and either three-, two-, or one-axis magnetometer data, although in the last case, an ambiguous solution, which cannot be resolved deterministically, is obtained. If attitude solutions are available, the spacecraft body rates are determined. Parameters to be displayed for attitude acquisition monitoring are calculated, and the requested attitude, attitude rate, and monitor parameters are then displayed.

In definitive mode, attitude solutions for IR/Sun, IR/magnetometer, and Sun/magnetometer data combinations are obtained whenever the appropriate sensor data are available. For combinations involving magnetometer data, attitude solutions for degraded (two-axis or one-axis) magnetometer data can be obtained. Magnetometer data for each axis is flagged as degraded if it has failed validation checks in the telemetry processor or data adjustment functions, or if the uncertainty in the associated residual bias is larger than some specified tolerance. For each attitude solution, the uncertainty resulting from both sensor errors and the basis geometry is calculated. The quality assurance parameter which indicates how many times a particular data combination provided an attitude solution is incremented, and both this number and the uncertainty are stored for use by the output and logging functions.

This definitive mode cycle is repeated until no data remains to be processed. A hardcopy printout of all attitude solutions and related uncertainties for each output time is produced when this cycle is completed. In interactive mode, these parameters are also displayed.

5.4.3.4 Differential Corrector Function

The differential corrector (DC) is employed whenever specified via NAMELIST or when the initializing function directs its use because of insufficient or poor quality sensor data. Any combination of IR scanner, Sun sensor, and magnetometer data and a batch mode differential corrector algorithm is used to estimate attitude propagation parameters and, optionally, magnetometer residual biases. The mathematical specification of this estimation process is given in Section 4.4.3.

Before processing, a message indicating that the differential correction method is being used is printed or displayed. After all data have been processed, the estimated value of each state vector element and related error statistics are printed or displayed. These state vector elements and error statistics may then be used as initial estimates for another application of the differential corrector. This cycle continues for either a user-specified maximum number of iterations or until the uncertainties for each state vector element are reduced to user-specified tolerances.

After this iteration procedure has been completed, the resulting estimates of the state vector elements and uncertainties are printed. In interactive mode, this information is displayed, and the operator may request continuation of this iteration procedure or a reexecution of the differential corrector with different values for the measurement weights, tolerances, or solve-for variables.

5.4.3.5 Smoothing Function

The smoothing function accepts attitude information from the discrete attitude determination or differential corrector functions, smooths the attitude solutions, and produces a final attitude solution (in the form of an Euler 2-1-3 pitch, roll, and yaw rotation) and associated uncertainty at user-specified times.

For data processed by the differential corrector, the final attitude solutions are generated by evaluating the attitude propagation polynomial (Equation (4-108)) whose coefficients are the attitude propagation parameters, at the user-requested times. Attitude uncertainties, based on Equation (4-121) are also evaluated.

When discrete attitude determination has been used, the multiple attitude solutions for each time, based on different data combinations, are combined to obtain an average attitude (Equation (4-136)) and associated uncertainty (Equations (4-140) through (4-142)) based on the weights attached to each attitude solution. These weights are calculated or, optionally, specified via NAMELIST. Optionally, the pitch, roll, and yaw angles are smoothed using a Chebyshev polynomial of user-specified maximum degree. If wheel desaturation occurred at any time during the pass, the smoothing option is not exercised. Otherwise, smoothing occurs whenever specified via NAMELIST or whenever a time gap in the attitude solutions requires interpolation. The goodness-of-fit parameter of the smoothing polynomial is calculated, and if it is larger than a user-specified tolerance, an option to flag attitude solutions which differ by more than a specified amount from the attitudes predicted from the polynomial and to reevaluate the polynomial using the remaining data is provided. In interactive mode, the operator may flag attitude solutions and request that smoothing be repeated using either a polynomial of different degree or different tolerances. When the smoothing process is complete, final attitude solutions of the user-requested times are computed from the best-fit Chebyshev polynomial.

When the final attitude solutions have been obtained, pitch, roll, and yaw rates are determined whenever the differential corrector was employed (Equation (4-116)), or whenever discrete attitude solutions were smoothed (Equation (4-146)). Pitch, roll, and yaw angle and rate quality assurance statistics (mean, rms, minimum, maximum) are calculated as required.

5.4.3.6 Output Function

After the final attitude and attitude rate solutions have been computed, various output parameters are calculated. Each pitch, roll, and yaw angle solution has an associated, one-byte flag value, and if the 3σ uncertainty for each solution is within the allowed tolerance (nominally 0.5 degree for pitch, 0.7 degree for roll, and 2.0 degrees for yaw), the flag byte is assigned the logical value "TRUE". Otherwise, the logical value "FALSE" is assigned. When all the flag bytes have been assigned, the percentage of the pitch, roll, and yaw data that have been flagged "FALSE" is calculated. Parameters indicating the quality of each type of discrete attitude solution are also computed.

5.4.4 Output From Attitude Determination Functions

The output from the attitude determination function during definitive operations consists of magnetometer biases and uncertainties for the magnetometer biases data set, attitude solutions and quality assurance parameters, and hardcopy printout. In acquisition mode, the output is restricted to hardcopy printout.

The magnetometer biases data set contains a history of the magnetometer residual biases and associated uncertainties. This information is used for validation of magnetometer data and for attitude determination. Whenever a valid estimate of the biases is obtained, the value of the residual bias on the X-, Y-, and Z-axis magnetometers, the uncertainties for each bias, and the start time of the pass for which estimates of the biases were made are recorded.

The attitude solution data consist of

- GMT of attitude solutions (seconds since zero hour UT September 1, 1957)
- Pitch, roll, and yaw angles (degrees)
- Pitch, roll, and yaw flags

The frequency of this data is user specified, approximately once every 10 seconds.

The quality assurance parameters include

- Pitch angle mean, rms, minimum, maximum, and uncertainty
- Roll angle mean, rms, minimum, maximum, and uncertainty
- Yaw angle mean, rms, minimum, maximum, and uncertainty

- Pitch rate mean, rms, minimum, maximum
- Roll rate mean, rms, minimum, maximum
- Yaw rate mean, rms, minimum, maximum
- Goodness-of-fit parameters for the smoothing polynominals for the pitch, roll, and yaw solutions
- Percentage of unacceptable pitch data (percentage of "FALSE" flag values)
- Percentage of unacceptable roll data (percentage of "FALSE" flag values)
- Percentage of unacceptable yaw data (percentage of "FALSE" flag values)
- Number of discrete scanner/Sun sensor solutions
- Average weight of scanner/Sun sensor solutions
- Number of discrete scanner/magnetometer solutions
- Average weight of scanner/magnetometer solutions
- Number of discrete scanner/degraded magnetometer solutions
- Average weight of scanner/degraded magnetometer solutions
- Number of discrete Sun sensor/magnetometer solutions
- Average weight of Sun sensor/magnetometer solutions
- Number of discrete Sun sensor/degraded magnetometer solutions
- Average weight of Sun sensor/degraded magnetometer solutions
- Maximum and minimum angular separation of nadir vector and Sun vector
- Maximum and minimum angular separation of nadir vector and magnetic field vector

- Maximum and minimum angular separation of Sun vector and magnetic field vector
- Flag indicating if valid magnetometer biases were determined
- Values of estimated magnetometer biases
- Uncertainties in the estimated magnetometer biases
- Value of constant pitch bias applied
- Value of constant roll bias applied
- Type of scanner pitch data used
- Type of scanner roll data used
- Scanner bias increment number from the scanner biases data set
- Attitude determination method (DISCRETE/RECURSIVE)

Printed output contains any of several levels of diagnostic printout, controllable through NAMELIST parameters, but it always contains a printout of the summary page. The contents of the summary page are

- Start time of pass (YYMMDD.HHMMSS)
- Stop time of pass (YYMMDD.HHMMSS)
- Start time of processing (YYMMDD.HHMMSS)
- Orbit number
- Station name
- Number of ephemeris tape used (if applicable)
- Mean pitch, roll, and yaw angles (degrees)
- Maximum magnitude of pitch, roll, and yaw angles (degrees)
- Mean pitch, roll, and yaw rates (degrees per second)

- Maximum magnitude of pitch, roll, and yaw rates (degrees per second)
- Number of attitude solutions
- Percentage of solutions unacceptable
- Longest gap in solutions (seconds)
- Percentage of valid magnetometer readings
- Percentage of valid Sun sensor readings
- Percentage of valid IR scanner readings
- Attitude history file created (YES, NO)
- Operator comments

5.4.5 Attitude Determination Displays

The attitude determination functions can optionally generate many displays for diagnostic purposes. Others are generated in normal processing, and these are described in the following subsections. Pertinent output data are produced in hardcopy; GESS displays need not be printed.

5.4.5.1 NAMELIST Display

The NAMELIST displays contain all parameters input through NAMELIST, which are listed in Section 5.4.2.

5.4.5.2 Attitude Determination Input Displays

Displays of all input data for the attitude determination functions are optionally available. These displays include

- Validated sensor data and ephemeris data from the data adjustment functions
- Quality assurance parameters generated by the data adjustment functions

- Current values of the scanner pitch and roll and the magnetometer biases
- Contents of the magnetometer biases data set

5.4.5.3 Attitude Determination Output Displays

In acquisition and orbit adjust modes, displays are provided to monitor the spacecraft acquisition and orbit adjust sequences. During mission mode, displays are provided to assist the operator in processing the data.

5.4.5.3.1 Acquisition Mode Displays

During acquisition mode, tabular displays and plots are provided to monitor attitude rate capture and movement of spacecraft pitch axis toward negative orbit normal. During the orbit adjust phase of the mission, displays are provided to monitor the transients in the attitude caused by the firing of the orbit adjust thruster.

These displays minimally include

- Pitch, roll, and yaw angles
- Pitch, roll, and yaw rates
- X-, Y-, and Z-components of the body rate vector
- Angle between the spacecraft Y-axis and the negative orbit normal (Equation (4-149))
- Y-axis attitude
- Magnitude and attitude of the spacecraft angular momentum vector
- Polynomials used to smooth the attitude and attitude rate solutions
- Phase plane plots (pitch/pitch rate, roll/roll rate, yaw/yaw rate)

5.4.5.3.2 Mission Mode Displays

During mission mode, displays are provided to assist the operator in processing the data. These displays include the results of various bias determination algorithms, attitude and attitude rate solutions, and quality assurance parameters. Parameters to be displayed include

- Pitch, roll, and yaw angles
- Pitch, roll, and yaw rates
- Standard deviations of the pitch, roll, and yaw angle solutions
- Quality assurance parameters
- Values of determined magnetometer biases and related uncertainties
- Method of attitude determination (discrete or differential corrector)
- Final values of the parameters solved for by the differential corrector and associated uncertainties

Both plots and tabular displays of the attitude solutions and standard deviations are provided.

Parameters which may optionally be displayed include

- Values of the parameters and associated uncertainties after each iteration of the differential corrector
- Angle between the Sun vector and nadir vector
- Angle between the Sun vector and magnetic field vector
- Angle between the nadir vector and magnetic field vector
- Spacecraft angular momentum vector attitude (right ascension and declination)
- Spacecraft nutation angle (angle between body rate vector and angular momentum vector)

Special displays are described in Figures 5-10 through 5-15.

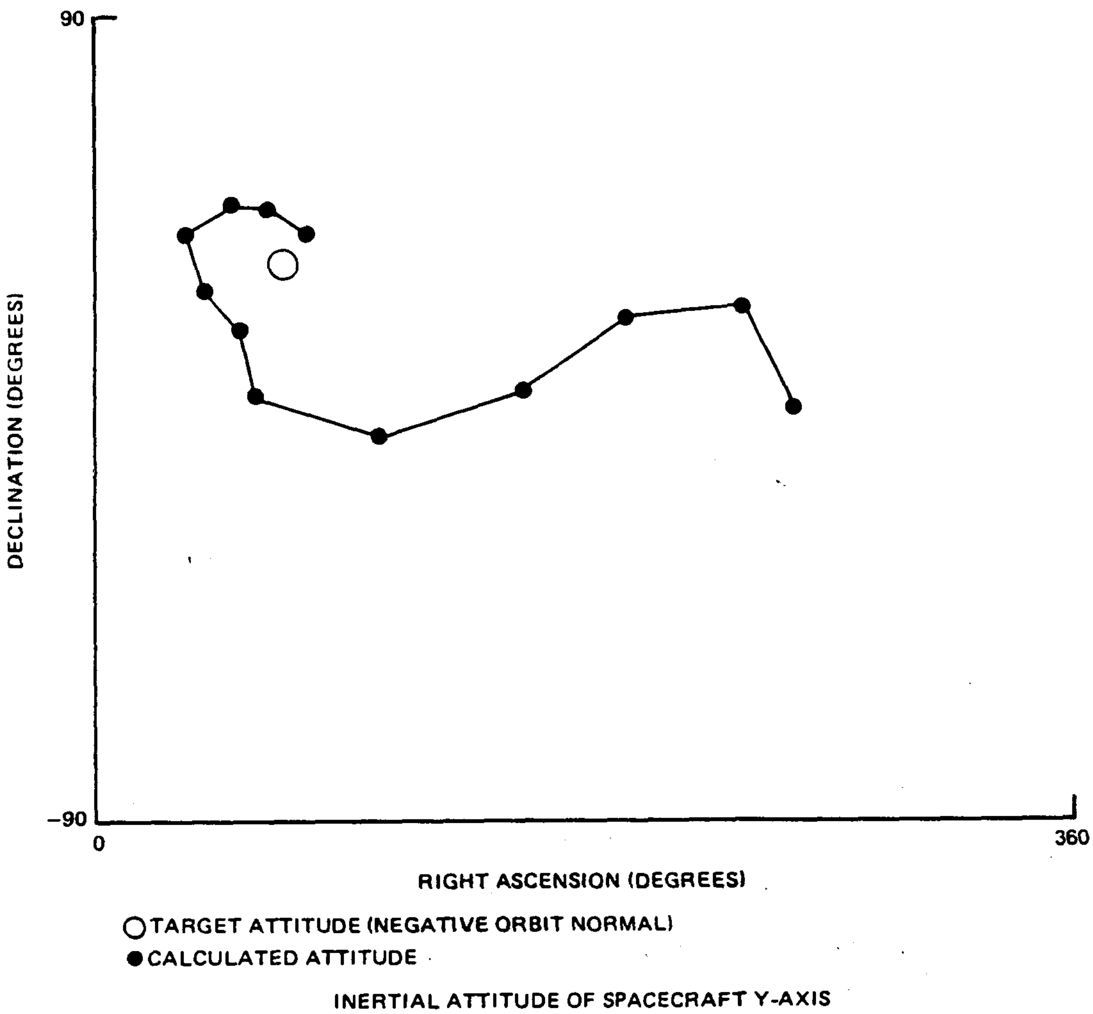


Figure 5-10. Spacecraft Y-Axis Attitude Vector Plot

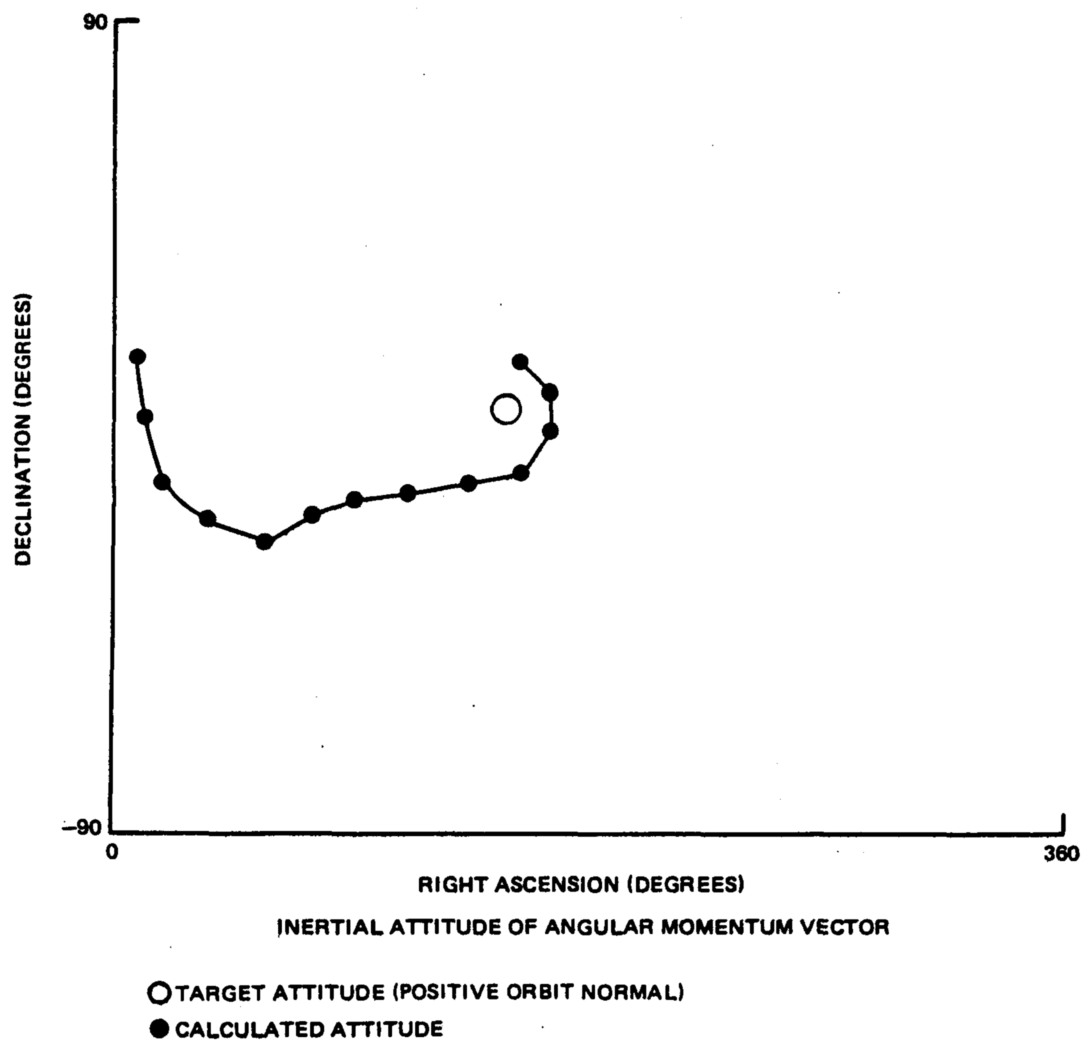


Figure 5-11. Angular Momentum Vector Attitude Plot

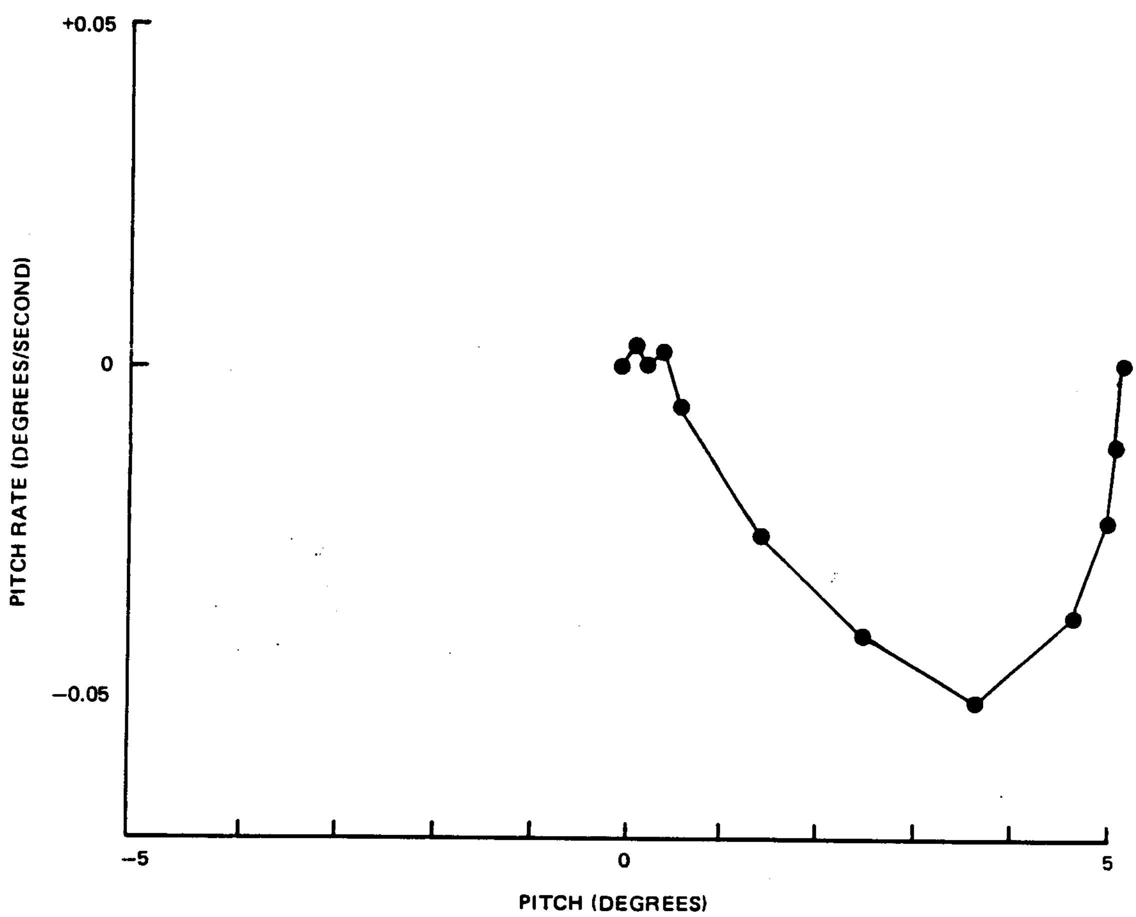


Figure 5-12. Pitch Capture Phase Plane Plot

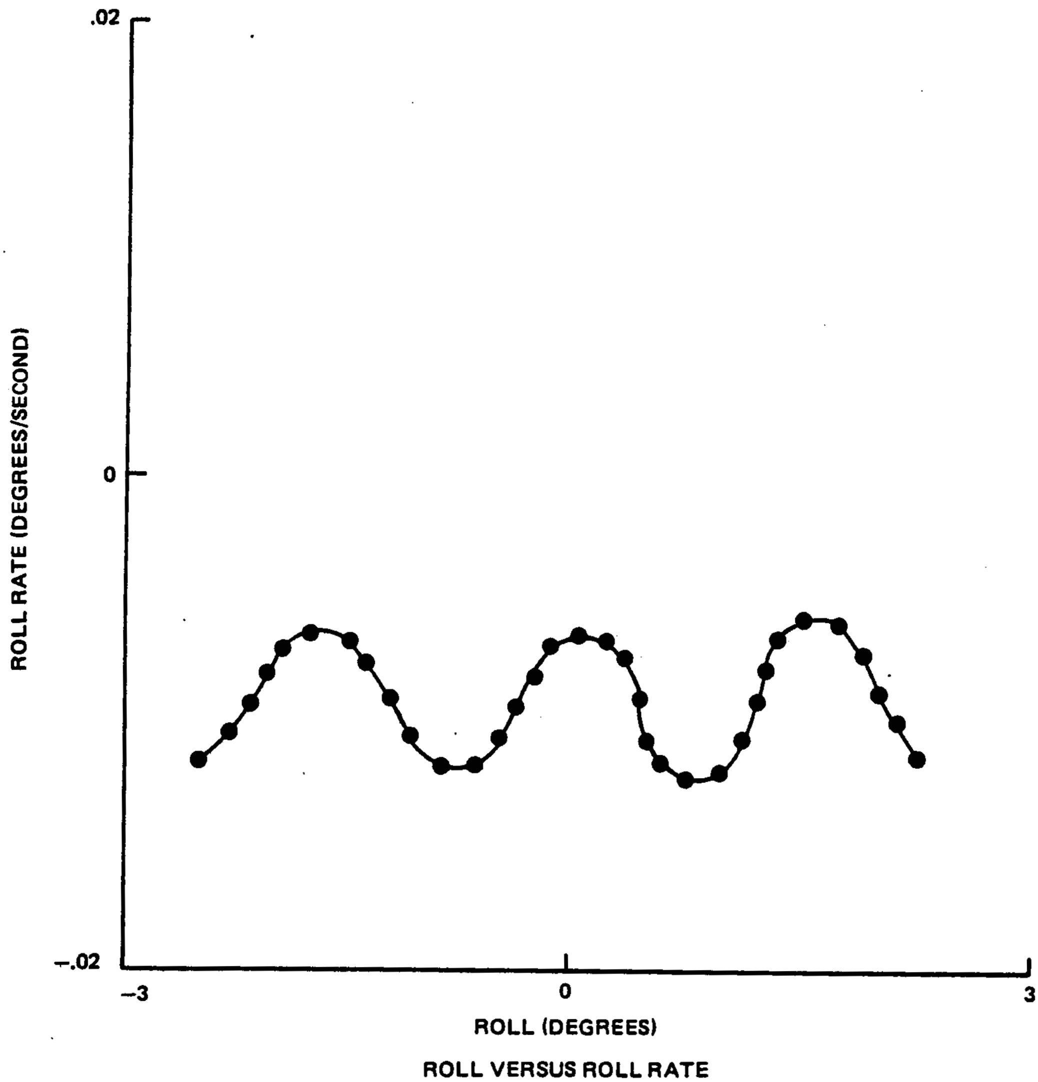


Figure 5-13. Roll Phase Plane Plot

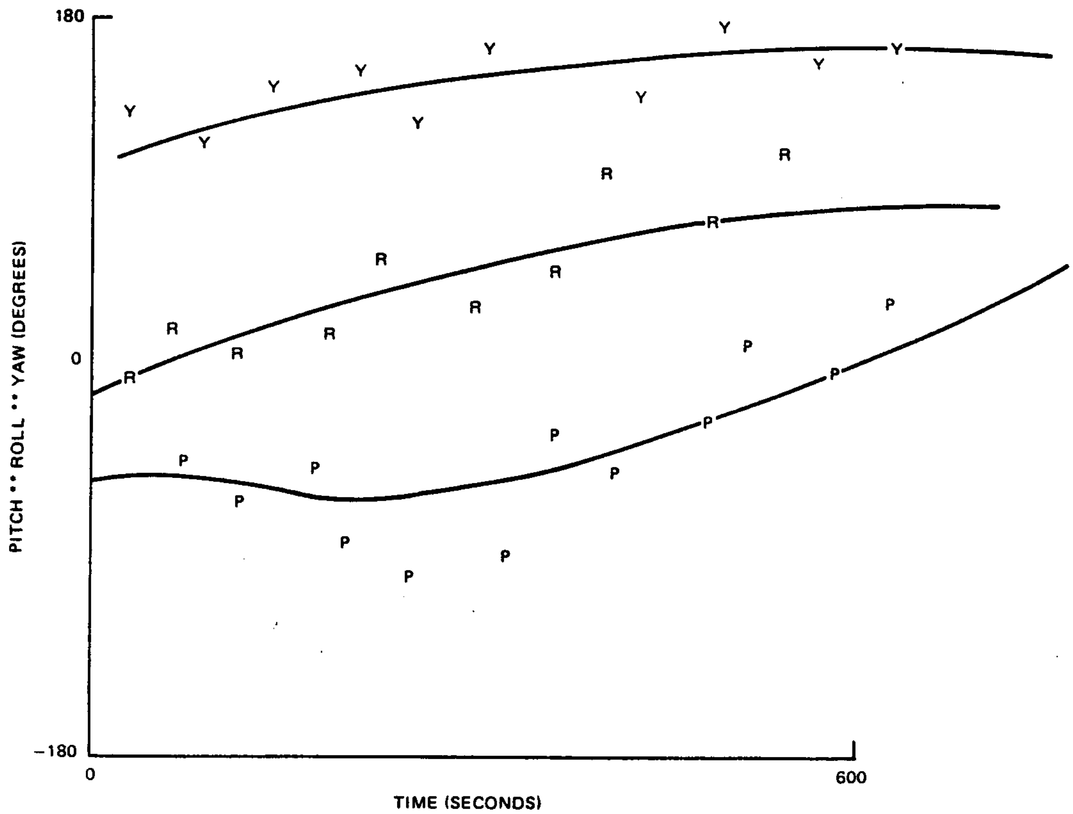


Figure 5-14. Raw and Smoothed Attitude Solutions Plot

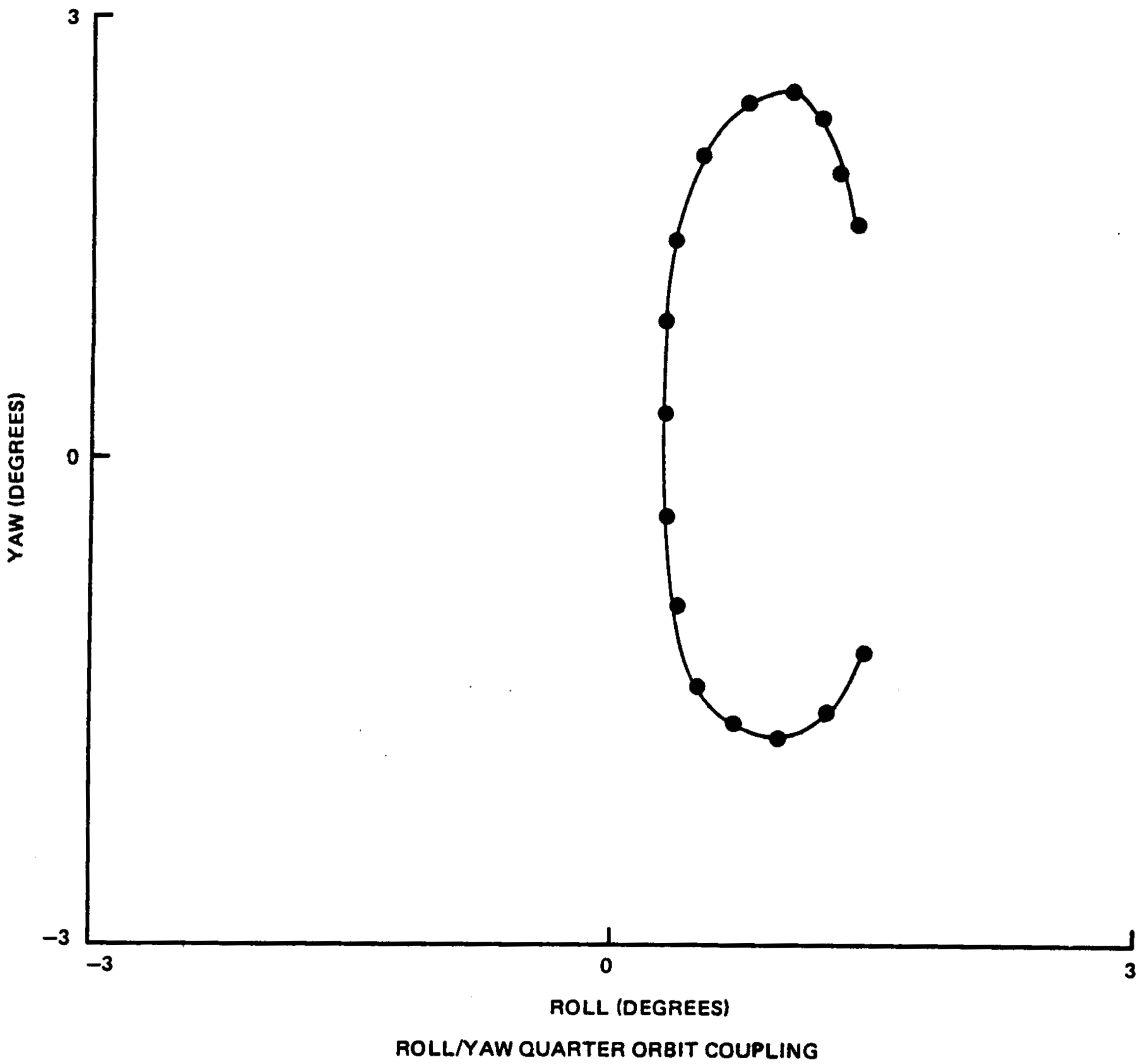


Figure 5-15. Quarter Orbit Coupling Plot

5.5 OUTPUT AND LOGGING FUNCTIONS

The output and logging functions include writing to two data sets. The first is the definitive attitude history file. Attitude solutions calculated by the attitude determination functions for each pass are entered into this partitioned data set as individual members. Member names are generated automatically and are always unique. The second data set is the log data set; parameters indicating the quality and quantity of sensor data and attitude solutions are written to this log. The functions are shown in Figure 5-16.

- WRITE ATTITUDE RESULTS TO ATTITUDE HISTORY FILE
- WRITE SUMMARY TO LOG DATA SET

Figure 5-16. Output and Logging Functions

5.5.1 Overview of the Output and Logging Functions

The output and logging functions include the following functions:

- Write the attitude solutions to the attitude history file data set
- Write a summary of these solutions to the log data set
- Verify that the data have been correctly written

5.5.2 Input to the Output and Logging Functions

The input consists of the 24-byte header information from the IPD telemetry header record, the attitude solutions generated by the attitude determination functions, quality assurance parameters generated by all the preceding functions, and NAMELIST parameters specifying whether or not to include the current solutions in the attitude history file. If the solutions are written to the history file, a summary of the solutions is also written into the log data set.

5.5.3 Description of the Output and Logging Functions

In interactive mode, the output and logging functions include displaying the quality assurance parameters which are included in the log data set summary. This display affords the operator three options:

1. To write the attitude solutions to the definitive attitude history file and the summary to the log data set

2. To reexecute the data adjustment or attitude determination functions

3. To write a summary to the log data set with a flag indicating which, if any, attitude solutions were not written to the definitive attitude history file due to poor quality

The attitude solutions are written provided that

1. The percentage of unacceptable pitch, roll, and yaw solutions must all be less than a NAMELIST parameter.

2. The largest gap in the discrete attitude solutions must be less than a NAMELIST parameter.

In addition, in interactive mode, the operator may determine whether or not to write the attitude solutions. If both criteria are satisfied, a unique member name is generated, the attitude solutions are written to the partitioned attitude history file, and the summary is written to the log data set. If either of these checks fails, the data adjustment and/or attitude determination functions may be reexecuted using different processing parameters.

If reprocessing does not produce attitude solutions of sufficient quality, nothing is written to the attitude history file, but a summary is written to the log data set indicating that the data were processed and no acceptable solutions were obtained. Hardcopy printout of the attitude solutions is generated.

The output and logging functions also include generating header information based on information appearing in the header record for IPD telemetry data or based on information appearing in the first telemetry record for MSOCC telemetry data. Generation of this header information includes automatic calculation of the date (re)processed; the operator is permitted to enter the number of times the pass of data has been reprocessed, and this value is included as part of the header information.

The header is placed in the first 24 bytes of the first record, and is followed by 64 bytes containing attitude solution quality information to be included in an IPD transmission header record. The DATA BASE USED value is set equal to the scanner bias increment number located in the scanner biases data set. These 88 bytes are followed by attitude solution data in a format compatible with IPD transmission data records. If more than one 3492-byte record is required for a set of data (which is unlikely for HCMM), subsequent data records for a pass contain only attitude solution data.

5.5.4 Output of the Output and Logging Functions

The output from the output and logging functions consists of

- Attitude solutions written to the attitude history file
- Attitude summaries written to the direct access log data set
- Hardcopy output indicating the quantity and quality of attitude solutions obtained, a flag indicating whether they were written to the history file, and a summary of the current status of the log data set

The attitude history file contains the spacecraft attitude (pitch, roll, and yaw angles), the associated times, and attitude solution quality flags.

5.5.5 Displays of the Output and Logging Functions

The input display for the output and logging functions minimally contains the following information:

- Mean (rms) pitch, roll, and yaw for each data pass
- Maximum and minimum pitch, roll, and yaw angle for each data pass
- Mean (rms) pitch, roll, and yaw rates for each data pass
- Maximum and minimum pitch, roll, and yaw rates for each data pass
- The time interval spanned by the attitude solutions

The output display minimally contains the following information:

- A field for operator comments
- The system-generated member name to be assigned to this segment of data in the attitude history file

5.6 DATA SET DEFINITIONS

Data sets required by the MSAD/HCMM system include

- NAMELIST data sets
- The telemetry data set
- A scanwheel speed data set
- An electromagnet data set
- A scanner bias utility source data set
- A scanner biases data set
- A magnetometer biases data set
- An attitude history file data set
- A log data set
- An attitude history transmission data set
- A horizon profile data set

These data sets are defined in subsequent subsections.

5.6.1 NAMELIST Data Sets

There are two NAMELIST data sets: one containing all four subsystem NAMELISTs and one containing the telemetry processing function unpacking and conversion parameters which are required for other than nominal processing. It is anticipated that the latter data set will not be required, but it is provided should nonnominal processing be requested. Both data sets have the following characteristics:

Size	Data Set Organization	DCB Parameters
One track	Sequential	Record Format = Fixed Blocked Logical Record Length = 80 Block Size = 7280

The contents of these data sets are described in Sections 5.2.2, 5.3.2, 5.4.2, and 5.5.2.

5.6.2 Telemetry Data Set

The ADL primary data set consists of 3520-byte records, each of which contains 1 IPD-generated 3492-byte record, or 11 OCC-generated 312-byte records. Appendix B contains memoranda of understanding from the ACA to MSOCC and IPD concerning the telemetry stream. The data set characteristics are

Size	Data Set Organization	DCB Parameters
TBD	**Direct access**	**Record Format = Fixed** Logical Record Length = 3520 Block Size = 3520

The contents of the data set are described in Figures 5-17 and 5-18.

5.6.3 Electromagnet Data Set

The electromagnet data set has the following characteristics:

Size	Data Set Organization	DCB Parameters
15 tracks	Direct access	Record Format = Fixed Blocked Logical Record Length = 20 Block Size = 7280

The electromagnet data set directory, or header records, consists of the first 50 logical records in the data set. The format is shown in Figure 5-18(a). The contents of the electromagnet data set are supplied at the telemetry data rate and are described in the following table:

Data Item	Format
Time (seconds since zero hours UT September 1, 1957)	R*8
Electromagnet, X-axis (pole-centimeters)	R*4
Electromagnet, Y-axis (pole-centimeters)	R*4
Electromagnet, Z-axis (pole-centimeters)	R*4

PROJECT TELEMETRY WORD IDENTIFIER	ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
	POCC HEADER 1	SPACECRAFT ID	1	1	0000 1000 (08 HEX)
	2	SPACECRAFT DATA MODE	2	1	0000 0000
	3	DATA TYPE	3	1	0000 0000
	4	TRANSMISSION RECORD NUMBER	4	2	BINARY INTEGER
	5	STATION NAME	6	3	FOUR XDS 930 CHARACTERS
	6	FILL	9	2	ALL ZEROS
	7	PASS CONTACT NUMBER	11	2	BINARY INTEGER
	8	FILL	13	4	ALL ZEROS
	9	AOS	17	4	YYDDDHHMM (BINARY INTEGER)
	10	FILL	21	4	ALL ZEROS
	11	DAY OF YEAR	25	2	BINARY INTEGER
	12	MILLISECONDS OF DAY	27	4	BINARY INTEGER
	13	QUALITY FLAG	31	1	0000 0000 = BAD 0000 0001 = GOOD
		TELEMETRY WORD			
A43	14	1	32	1	SCANWHEEL SPEED (COARSE)
A1	15	4	33	1	MAGNETOMETER ROLL 1
A2	16	5	34	1	MAGNETOMETER PITCH 1
A3	17	6	35	1	MAGNETOMETER YAW 1
A27	18	11	36	1	ROLL ERROR (FINE)
T2	19	13	37	1	MINOR FRAME COUNTER
T7	20	14–16	38	3	SPACECRAFT CLOCK

Figure 5-17. Contents of MSOCC Telemetry Records (1 of 3)

PROJECT TELEMETRY WORD IDENTIFIER	ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
		TELEMETRY WORD			
A5	21	17	41	1	ELECTROMAGNET BIAS (X-AXIS)
A6	22	18	42	1	ELECTROMAGNET BIAS (Y-AXIS)
A7	23	20	43	1	ELECTROMAGNET BIAS (Z-AXIS)
A8	24	21	44	1	ELECTROMAGNET MOMENT (ROLL)
A9	25	33	45	1	ELECTROMAGNET MOMENT (PITCH)
A10	26	34	46	1	ELECTROMAGNET MOMENT (YAW)
A1	27	36	47	1	MAGNETOMETER ROLL 2
A2	28	37	48	1	MAGNETOMETER PITCH 2
A3	29	38	49	1	MAGNETOMETER YAW 2
A11	30	40	50	1	ROLL ERROR (COARSE)
A12	31	41	51	1	PITCH ERROR (COARSE)
A14	32	42	52	1	SCANWHEEL SPEED (FINE)
A15	33	43	53	1	ROLL BIAS
T13	34	48	54	1	PCM MODE WORD
T3	35	53	55	1	A/D CONV CAL LEVEL 1
T4	36	54	56	1	A/D CONV CAL LEVEL 2
T5	37	56	57	1	A/D CONV CAL LEVEL 3
BL10	38	58	58	1	SUN SENSOR ID AND STATUS FLAGS
BL11	39	59	59	1	N_a (SUN SENSOR)
BL12	40	60	60	1	N_b (SUN SENSOR)
A1	41	68	61	1	MAGNETOMETER ROLL 3

Figure 5-17. Contents of MSOCC Telemetry Records (2 of 3)

PROJECT TELEMETRY WORD IDENTIFIER	ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
		TELEMETRY WORD			
A2	42	69	62	1	MAGNETOMETER PITCH 3
A3	43	70	63	1	MAGNETOMETER YAW 3
SC1	44	77	64	1	STATUS FLAGS
A16	45	93	65	1	5-VOLT SUPPLY
A1	46	100	66	1	MAGNETOMETER ROLL 4
A2	47	101	67	1	MAGNETOMETER PITCH 4
A3	48	102	68	1	MAGNETOMETER YAW 4
A21	49	112	69	1	MAGNETOMETER FIELD RATE (ROLL)
A22	50	116	70	1	MAGNETOMETER FIELD RATE (PITCH)
A23	51	117	71	1	MAGNETOMETER FIELD RATE (YAW)
A24	52	118	72	1	SCANWHEEL DUTY CYCLE I
A28	53	120	73	1	PITCH ERROR (FINE)
A29	54	122	74	1	SCANWHEEL DUTY CYCLE II
A41	55	125	75	1	10-VOLT SUPPLY
	56	FILL	76	9	ALL ZEROS
	57	DAY OF YEAR	85	2	NEXT MINOR FRAME
	96	FILL	130	9	
	97	DAY OF YEAR	145	2	NEXT MINOR FRAME
	136	FILL	190	9	
	137	DAY OF YEAR	205	2	NEXT MINOR FRAME
	176	FILL	250	9	
	177	FILL	265	48	ALL ZEROS
				312 BYTES IN RECORD	

Figure 5-17. Contents of MSOCC Telemetry Records (3 of 3)

PROJECT TELEMETRY WORD IDENTIFIER	ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
	IPD HEADER 1	SPACECRAFT IDENTIFICATION	1	1	1010 1000 (A8 HEX)
	2	SPACECRAFT DATA MODE	2	1	0000 0000
	3	DATA TYPE	3	1	0000 0001
	4	RUNNING NUMBER OF RECORDS/ BLOCK (PASS)	4	2	BINARY INTEGER
	5	STATION NAME	6	3	FOUR XDS930 CHARACTERS
	6	RUNNING NUMBER OF RECORDS IN TRANSMISSION	9	2	BINARY INTEGER
	7	PASS CONTACT NUMBER (ORBIT NUMBER)	11	2	BINARY INTEGER
	8	DATE PROCESSED AND NUMBER OF TIMES REPROCESSED/IPD	13	4	BINARY INTEGER
	9	START TIME OF BLOCK (PASS)	17	4	YYDDDHHMM (BINARY INTEGER)
	10	STOP TIME OF BLOCK (PASS)	21	4	YYDDDHHMM (BINARY INTEGER)
	11	DAY OF YEAR	25	2	BINARY INTEGER
	12	MILLISECONDS OF DAY	27	4	BINARY INTEGER
	13	QUALITY FLAG	31	1	0000 0000 = BAD 0000 0001 = GOOD
		TELEMETRY WORD			
A43	14	1	32	1	SCANWHEEL SPEED (COARSE)
A1	15	4	33	1	MAGNETOMETER ROLL 1
A2	16	5	34	1	MAGNETOMETER PITCH 1
A3	17	6	35	1	MAGNETOMETER YAW 1
A27	18	11	36	1	ROLL ERROR (FINE)

Figure 5-18. Contents of IPD Telemetry Records (1 of 3)

PROJECT TELEMETRY WORD IDENTIFIER	ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
		TELEMETRY WORD			
T2	19	13	37	1	MINOR FRAME COUNTER
T7	20	14–16	38	3	SPACECRAFT CLOCK
A5	21	17	41	1	ELECTROMAGNET BIAS (X–AXIS)
A6	22	18	42	1	ELECTROMAGNET BIAS (Y–AXIS)
A7	23	20	43	1	ELECTROMAGNET BIAS (Z–AXIS)
A8	24	21	44	1	ELECTROMAGNET MOMENT (ROLL)
A9	25	33	45	1	ELECTROMAGNET MOMENT (PITCH)
A10	26	34	46	1	ELECTROMAGNET MOMENT (YAW)
A1	27	36	47	1	MAGNETOMETER ROLL 2
A2	28	37	48	1	MAGNETOMETER PITCH 2
A3	29	38	49	1	MAGNETOMETER YAW 2
A11	30	40	50	1	ROLL ERROR (COARSE)
A12	31	41	51	1	PITCH ERROR (COARSE)
A14	32	42	52	1	SCAN WHEEL SPEED (FINE)
A15	33	43	53	1	ROLL BIAS
T13	34	48	54	1	PCM MODE WORD
T3	35	53	55	1	A/D CONV CAL LEVEL 1
T4	36	54	56	1	A/D CONV CAL LEVEL 2
T5	37	56	57	1	A/D CONV CAL LEVEL 3
BL10	38	58	58	1	SUN SENSOR ID AND STATUS FLAGS
BL11	39	59	59	1	N_a (SUN SENSOR)

Figure 5-18. Contents of IPD Telemetry Records (2 of 3)

PROJECT TELEMETRY WORD IDENTIFIER	ITEM NUMBER	LABEL	LOCATON (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
		TELEMETRY WORD			
BL12	40	60	60	1	N_b (SUN SENSOR)
A1	41	68	61	1	MAGNETOMETER ROLL 3
A2	42	69	62	1	MAGNETOMETER PITCH 3
A3	43	70	63	1	MAGNETOMETER YAW 3
SC1	44	77	64	1	STATUS FLAGS
A26	45	93	65	1	5-VOLT SUPPLY
A1	46	100	66	1	MAGNETOMETER ROLL 4
A2	47	101	67	1	MAGNETOMETER PITCH 4
A3	48	102	68	1	MAGNETOMETER YAW 4
A21	49	112	69	1	MAGNETOMETER FIELD RATE (ROLL)
A22	50	116	70	1	MAGNETOMETER FIELD RATE (PITCH)
A23	51	117	71	1	MAGNETOMETER FIELD RATE (YAW)
A24	52	118	72	1	SCANWHEEL DUTY CYCLE I
A28	53	120	73	1	PITCH ERROR (FINE)
A29	54	122	74	1	SCANWHEEL DUTY CYCLE II
A41	55	125	76	1	10-VOLT SUPPLY
	56	FILL	73	9	ALL ZEROS
	57	DAY OF YEAR	85	2	55 REPETITIONS OF ITEMS 11 THROUGH 56
	2530	FILL	3375	9	
	2531	FILL	3385	108	ALL ZEROS
				3492 BYTES IN RECORD	

Figure 5-18. Contents of IPD Telemetry Records (3 of 3)

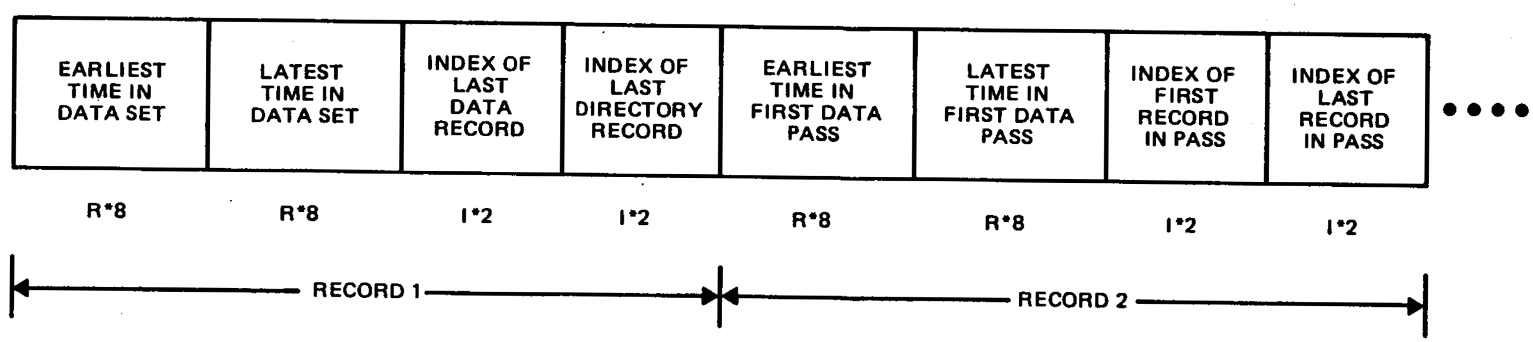


Figure 5-18(a). Electromagnet Data Set Header Record Format

5.6.4 Scanwheel Speed Data Set

The scanwheel speed data set has the following characteristics:

Size	Data Set Organization	DCB Parameters
10 tracks	Direct access	Record Format = Fixed Blocked Logical Record Length = 12 Block Size = 7284

The scanwheel speed data set directory, or header records, consists of the first 100 records. The format is shown in Figure 5-18(b). The contents of the scanwheel speed data set are generated at the telemetry data rate and are described in the following table:

Data Item	Format
Time (seconds since zero hours UT September 1, 1957)	R*8
Scanwheel speed (rpm)	R*4

5.6.5 Scanner Bias Utility Source Data Set

The characteristics of the scanwheel bias utility source data set are

Size	Data Set Organization	DCB Parameters
20 tracks	Sequential	Record Format = Fixed Blocked Logical Record Length = 48 Block Size = 3360

The contents of the scanwheel bias utility source data set are generated at the attitude solution period (nominally every 10 seconds) and are described in the following table:

Data Item	Format
Time (YYMMDD. HHMMSS)	R*8
Unit Sun vector in body coordinates (X-axis)	R*4

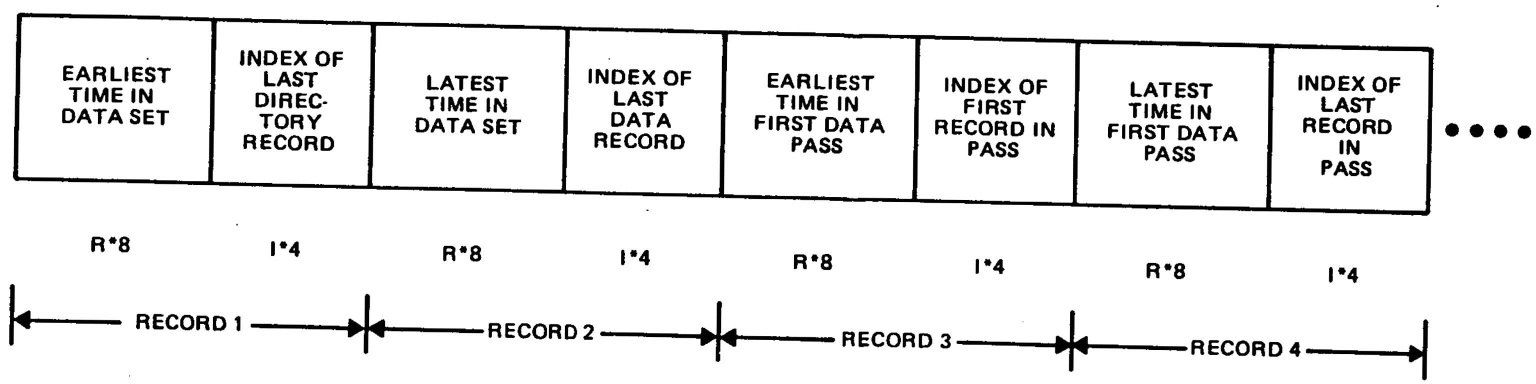


Figure 5-18(b). Scanwheel Speed Data Set Header Record Format

Data Item	Format
Unit Sun vector in body coordinates (Y-axis)	R*4
Unit Sun vector in body coordinates (Z-axis)	R*4
Unit Sun vector in orbital coordinates (Z-axis)	R*4
Scanner fine pitch angle (degrees)	R*4
Scanner coarse pitch angle (degrees)	R*4
Scanner duty cycle-generated pitch angle (degrees)	R*4
Scanner fine roll angle (degrees)	R*4
Scanner coarse roll angle (degrees)	R*4
Scanner duty cycle-generated roll angle (degrees)	R*4

The data set disposition shall be SHR in the job control language and changed to MOD at execution time to permit data concatenation.

5.6.6 Scanner Biases Data Set

The characteristics of the scanner biases data set are

Size	Data Set Organization	DCB Parameters
One track	Sequential	Record Format = Fixed Blocked Logical Record Length = 69 Block Size = 69

The contents of the scanner biases data set are described in the following table:

Data Item	Format
Start time of pass (YYMMDD.HHMMSS)	R*8
Stop time of pass (YYMMDD.HHMMSS)	R*8
Date generated (YYMMDD)	R*4
Estimate of fine pitch bias (degrees)	R*4
Estimate of coarse pitch bias (degrees)	R*4

Data Item	Format
Estimate of duty cycle-generated pitch bias (degrees)	R*4
Estimate of fine roll bias (degrees)	R*4
Estimate of coarse roll bias (degrees)	R*4
Estimate of duty cycle-generated roll bias (degrees)	R*4
Standard deviation of fine pitch bias (degrees)	R*4
Standard deviation of coarse pitch bias (degrees)	R*4
Standard deviation of duty cycle-generated pitch bias (degrees)	R*4
Standard deviation of fine roll bias (degrees)	R*4
Standard deviation of coarse roll bias (degrees)	R*4
Standard deviation of duty cycle-generated roll bias (degrees)	R*4
Scanner bias increment number	L*1

5.6.7 Magnetometer Biases Data Set

The characteristics of the magnetometer biases data set are

Size	Data Set Organization	DCB Parameters
One track	Direct access	Record Format = Fixed Blocked Logical Record Length = 40 Block Size = 40

The contents of the magnetometer biases data set are generated on option every sunlit pass and are described in the following table:

Data Item	Format
Start time of pass (YYMMDD.HHMMSS)	R*8
Stop time of pass (YYMMDD.HHMMSS)	R*8
Magnetometer bias (X-axis) (millioersteds)	R*4
Magnetometer bias (Y-axis) (millioersteds)	R*4

Data Item	Format
Magnetometer bias (Z-axis) (millioersteds)	R*4
Standard deviation of magnetometer bias (X-axis) (millioersteds)	R*4
Standard deviation of magnetometer bias (Y-axis) (millioersteds)	R*4
Standard deviation of magnetometer bias (Z-axis) (millioersteds)	R*4

5.6.8 Attitude History File Data Set

Appendix B references a memorandum of understanding from the ACA to IPD concerning the disposition of the definitive attitude solutions.

The characteristics of the attitude history file data set are

Size	Data Set Organization	DCB Parameters
TBD	Partitioned	Record Fromat = Fixed Blocked Logical Record Length = 3492 Block Size = 3492

The contents of each member of the attitude history file data set are described in Figure 5-19.

5.6.9 Log Data Set

The characteristics of the log data set are

Size	Data Set Organization	DCB Parameters
TBD	Direct access	Record Format = Fixed Blocked Logical Record Length = 404 Block Size = 404

ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
1	SPACECRAFT IDENTIFICATION	1	1	1010 1000
2	SPACECRAFT DATA MODE	2	1	0000 0000
3	DATA TYPE	3	1	0000 0010
4	RECORDS/PASS	4	2	BINARY INTEGER
5	FILL	6	3	ZEROS
6	RECORDS/TRANSMISSION	9	2	BINARY INTEGER
7	FILL	11	2	ZEROS
8	DATE PROCESSED AND NUMBER OF TIMES	13	4	YYDDDNN (I*4)
9	START TIME OF PASS	17	4	YYDDDHHMM (I*4)
10	STOP TIME OF PASS	21	4	YYDDDHHMM (I*4)
11	DATA BASE USED	25	4	I*4
12	OUTPUT FREQUENCY (SECONDS)	29	4	I*4
13	MAXIMUM PITCH (DEGREES)	33	4	R*4
14	MEAN PITCH (DEGREES)	37	4	R*4
15	MINIMUM PITCH (DEGREES)	41	4	R*4
16	MAXIMUM ROLL (DEGREES)	45	4	R*4
17	MEAN ROLL (DEGREES)	49	4	R*4
18	MINIMUM ROLL (DEGREES)	53	4	R*4
19	MAXIMUM YAW (DEGREES)	57	4	R*4
20	MEAN YAW (DEGREES)	61	4	R*4
21	MINIMUM YAW (DEGREES)	65	4	R*4
22	PERCENT UNACCEPTABLE PITCH	69	4	R*4

HEADER

Figure 5-19. Contents of Each Member of the Attitude History File Data Set (1 of 2)

ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
23	PERCENT UNACCEPTABLE ROLL	73	4	R*4
24	PERCENT UNACCEPTABLE YAW	77	4	R*4
25	START TIME	81	4	YYDDDHHMM (I*4)
26	STOP TIME	85	4	YYDDDHHMM (I*4)
27	DAY OF YEAR	89	2	I*2
28	MILLISECOND OF DAY	91	4	I*4
29	PITCH FLAG	95	1	L*1 .TRUE. = FLAGGED OR BAD DATA
30	ROLL FLAG	96	1	L*1 .TRUE. = FLAGGED OR BAD DATA
31	YAW FLAG	97	1	L*1 .TRUE. = FLAGGED OR BAD DATA
32	FILL	98	3	ZEROS
33	PITCH (DEGREES)	101	4	R*4
34	ROLL (DEGREES)	105	4	R*4
35	YAW (DEGREES)	109	4	R*4
36 • • •	DAY OF YEAR • • •	113 • • •	2 • • •	REPEAT ITEMS 27–35 AS NEEDED; PAD WITH ZERO FILL AT END

Figure 5-19. Contents of Each Member of the Attitude History File Data Set (2 of 2)

The contents of the log data set are generated once per pass of data, on option, and are described in the following table:

Data Item	Format
Start time of data being summarized (YYMMDD.HHMMSS)	R*8
Stop time of data being summarized (YYMMDD.HHMMSS)	R*8
Orbit number	I*2
Station receiving telemetry data	I*4
Mean pitch angle during the pass (degrees)	R*4
Mean roll angle during the pass (degrees)	R*4
Mean yaw angle during the pass (degrees)	R*4
Rms pitch angle (degrees)	R*4
Rms roll angle (degrees)	R*4
Rms yaw angle (degrees)	R*4
Standard deviation of the mean pitch angle (degrees)	R*4
Standard deviation of the mean roll angle (degrees)	R*4
Standard deviation of the mean yaw angle (degrees)	R*4
Maximum value of pitch during the pass (degrees)	R*4
Maximum value of roll during the pass (degrees)	R*4
Maximum value of yaw during the pass (degrees)	R*4
Minimum value of pitch during the pass (degrees)	R*4
Minimum value of roll during the pass (degrees)	R*4
Minimum value of yaw during the pass (degrees)	R*4
Mean pitch rate during the pass (degrees per second)	R*4
Mean roll rate during the pass (degrees per second)	R*4

Data Item	Format
Mean yaw rate during the pass (degrees per second)	R*4
Rms pitch rate (degrees per second)	R*4
Rms roll rate (degrees per second)	R*4
Rms yaw rate (degrees per second)	R*4
Minimum pitch rate during the pass (degrees per second)	R*4
Minimum roll rate during the pass (degrees per second)	R*4
Minimum yaw rate during the pass (degrees per second)	R*4
Maximum pitch rate during the pass (degrees) per second)	R*4
Maximum roll rate during the pass (degrees per second)	R*4
Maximum yaw rate during the pass (degrees per second)	R*4
Goodness-of-fit parameter for pitch smoothing polynomial	R*4
Goodness-of-fit parameter for roll smoothing polynomial	R*4
Goodness-of-fit parameter for yaw smoothing polynomial	R*4
Percentage unacceptable pitch data (percentage of "FALSE" flag values)	R*4
Percentage unacceptable roll data (percentage of "FALSE" flag values)	R*4
Percentage unacceptable yaw data (percentage of "FALSE" flag values)	R*4
Percentage acceptable attached GMT data per pass (0-100)	I*2
Scanwheel desaturation flag (0 or 1)	L*1
Percentage acceptable scanner pitch data after data adjustment tests (0-100)	I*2

Data Item	Format
Percentage acceptable scanner roll data after data adjustment tests (0-100)	I*2
Percentage of acceptable Sun sensor data after adjustment tests (0-100)	I*2
Percentage of acceptable magnetometer data after data adjustment tests (0-100)	I*2
Quality of the valid scanner data (rms from null) (degrees)	R*4
Quality of valid Sun sensor data (rms from null) (degrees)	R*4
Quality of valid magnetometer data (rms from null) (degrees)	R*4
Rms difference from nominal for magnetic field magnitude (millioersteds)	R*4
Quality of valid Sun/magnetometer data from dot product check (degrees)	R*4
Quality of valid Sun/scanner data from dot product check (degrees)	R*4
Quality of valid magnetometer/scanner data from dot product check (degrees)	R*4
Number of discrete scanner/Sun sensor solutions	I*2
Average weight of scanner/Sun sensor pitch solutions	R*4
Average weight of scanner/Sun sensor roll solutions	R*4
Average weight of scanner/Sun sensor yaw solutions	R*4
Number of discrete scanner/magnetometer solutions	I*2
Average weight of scanner/magnetometer pitch solutions	R*4
Average weight of scanner/magnetometer roll solutions	R*4
Average weight of scanner/magnetometer yaw solutions	R*4

Data Item	Format
Number of discrete scanner/degraded magnetometer solutions	I*2
Average weight of scanner/degraded magnetometer pitch solutions	R*4
Average weight of scanner/degraded magnetometer roll solutions	R*4
Average weight of scanner/degraded magnetometer yaw solutions	R*4
Number of discrete Sun sensor/magnetometer solutions	I*2
Average weight of Sun sensor/magnetometer pitch solutions	R*4
Average weight of Sun sensor/magnetometer roll solutions	R*4
Average weight of Sun sensor/magnetometer yaw solutions	R*4
Number of discrete Sun sensor/degraded magnetometer solutions	I*2
Average weight of Sun sensor/degraded magnetometer pitch solutions	R*4
Average weight of Sun sensor/degraded magnetometer roll solutions	R*4
Average weight of Sun sensor/degraded magnetometer yaw solutions	R*4
Minimum angular separation of nadir and Sun vectors (degrees)	R*4
Maximum angular separation of nadir and Sun vectors (degrees)	R*4
Minimum angular separation of Sun and magnetic field vectors (degrees)	R*4
Maximum angular separation of Sun and magnetic field vectors (degrees)	R*4
Flag indication of whether magnetometer biases were determined.	L*1

Data Item	Format
Estimated magnetometer bias (X-axis) (millioersteds)	R*4
Estimated magnetometer bias (Y-axis) (millioersteds)	R*4
Estimated magnetometer bias (Z-axis) (millioersteds)	R*4
Standard deviation in estimated magnetometer bias (X-axis) (millioersteds)	R*4
Standard deviation in estimated magnetometer bias (Y-axis) (millioersteds)	R*4
Standard deviation in estimated magnetometer bias (Z-axis) (millioersteds)	R*4
Value of constant pitch bias applied (degrees)	R*4
Value of constant roll bias applied (degrees)	R*4
Largest gap in discrete attitude solutions (seconds)	I*2
Scanner bias increment number	L*1
Type of attitude determination method	I*4
Type of scanner pitch data used	L*1
Type of scanner roll data used	L*1
Member name of corresponding attitude solutions in attitude history file	R*8

Data Item	Format
Wall clock time when processed	R*8
Date processed	R*4
Operator's initials	L*4
Status flag	L*1
Operator comment field	L*1(20)
Analyst's initials	L*4
Date analyst reviewed summary	R*4
Initials of operator who added data to transmission data set	L*4
Date added to transmission data set	R*4
Initials of operator who confirmed data transmitted to IPD	L*4
Date transmission to IPD confirmed	R*4
Backup tape number	R*8
Backup tape file number	I*2
Fill	L*1(10)

5.6.10 Attitude Solution Transmission Data Set

The characteristics of the attitude solution transmission data set are

Size	Data Set Organization	DCB Parameters
TBD	Sequential	Record Format = Fixed Blocked Logical Record Length = 3492 Block Size = 3492

The attitude solution transmission data set consists of concatenated segments of data from the attitude history file. Each segment (pass) contains a header record as shown in Figure 5-20. The attitude solution data records are shown in Figure 5-21.

	ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
HEADER	1	SPACECRAFT IDENTIFICATION	1	1	1010 1000
HEADER	2	DATA MODE	2	1	0000 0000
HEADER	3	DATA TYPE	3	1	0000 0010
HEADER	4	RECORDS/PASS	4	2	BINARY INTEGER
HEADER	5	FILL	6	3	BINARY INTEGER
HEADER	6	RECORDS/TRANSMISSION	9	2	BINARY INTEGER
HEADER	7	FILL	11	2	ZEROS
HEADER	8	DATE PROCESSED AND NUMBER OF TIMES REPROCESSED	13	4	YYDDDNN
HEADER	9	START TIME OF PASS	17	4	YYDDDHHMM
HEADER	10	STOP TIME OF PASS	21	4	YYDDDHHMM
	11	DATA BASE USED	25	4	I*4
	12	OUTPUT FREQUENCY (SECONDS)	29	4	I*4
	13	MAXIMUM PITCH (DEGREES)	33	4	R*4
	14	MEAN PITCH (DEGREES)	37	4	R*4
	15	MINIMUM PITCH (DEGREES)	41	4	R*4
	16	MAXIMUM ROLL (DEGREES)	45	4	R*4
	17	MEAN ROLL (DEGREES)	49	4	R*4
	18	MINIMUM ROLL (DEGREES)	53	4	R*4
	19	MAXIMUM YAW (DEGREES)	57	4	R*4
	20	MEAN YAW (DEGREES)	61	4	R*4

Figure 5-20. Transmission Data Set Header Record (1 of 2)

ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
21	MINIMUM YAW (DEGREES)	65	4	R*4
22	PERCENT UNACCEPTABLE PITCH	69	4	R*4
23	PERCENT UNACCEPTABLE ROLL	73	4	R*4
24	PERCENT UNACCEPTABLE YAW	77	4	R*4
25	START TIME	81	4	YYDDDHHMM
26	STOP TIME	85	4	YYDDDHHMM
27	FILL	89	3404	ZEROS
			3492 BYTES	

Figure 5-20. Transmission Data Set Header Record (2 of 2)

ITEM NUMBER	LABEL	LOCATION (BYTE NUMBER)	LENGTH (BYTES)	CONTENTS
1	SPACECRAFT IDENTIFICATION	1	1	1010 1000
2	DATA MODE	2	1	0000 0000
3	DATA TYPE	3	1	0000 0010
4	RECORDS/PASS	4	2	BINARY INTEGER
5	FILL	6	3	ZEROS
6	RECORDS/TRANSMISSION	9	2	BINARY INTEGER
7	FILL	11	2	ZEROS
8	DATE PROCESSED AND NUMBER OF TIMES REPROCESSED	13	4	YYDDDNN
9	START TIME OF PASS	17	4	YYDDDHHMM
10	STOP TIME OF PASS	21	4	YYDDDHHMM
11	DAY OF YEAR	25	2	I*2
12	MILLISECONDS OF DAY	27	4	I*4
13	PITCH FLAG	31	1	L*1
14	ROLL FLAG	32	1	L*1
15	YAW FLAG	33	1	L*1
16	FILL	34	3	ZEROS
17	PITCH (DEGREES)	37	4	R*4
18	ROLL (DEGREES)	41	4	R*4
19	YAW (DEGREES)	45	4	R*4
20	DAY OF YEAR	49	2	} REPEAT ITEMS 11-19 AS NEEDED (MAXIMUM OF 144 SETS OF DATA POINTS PER RECORD) AND PAD WITH ZERO FILL AT END

Figure 5-21. Transmission Data Set Data Record

5.6.11 Horizon Altitude Data Set

The horizon altitude data set will be written using data obtained from the infrared horizon radiance modeling (HRM) utility and expanded by a utility program that interpolates this data to intervals of 15 days and 2 degrees of subsatellite point latitude. The characteristics of this data set are

Size	Data Set Organization	DCB Parameters
10 tracks	Sequential	Record Format = Fixed Logical Record Length = 20 Block Size = 2000

The contents and organization of the data set are illustrated in Figure 5-22. The attitude determination system will perform a linear interpolation on this data to obtain the height of the CO_2 layer at AOS, $h_I(\lambda_S, t)$, and the height of the CO_2 layer at LOS, $h_O(\lambda_S, t)$, for the given day and subsatellite point latitude, λ_S. Note that these CO_2 layer heights are different for northbound and southbound portions of the orbit.

The characteristics of the horizon altitude data set are

Item Number	Data Item	Data Type and Length
1	Day of year (= 1)	I*2
2	Latitude of subsatellite point (= 84 degrees)	I*2
3	CO_2 height (kilometers), AOS, southbound	R*4
4	CO_2 height (kilometers), LOS, southbound	R*4
5	CO_2 height (kilometers), AOS, northbound	R*4
6	CO_2 height (kilometers), LOS, northbound	R*4
7	Day of year (= 1)	I*2
8	Latitude of subsatellite point (= 82 degrees)	I*2
9–12	Same as items 3 through 6	4 (R*4)

Item Number	Data Item	Data Type and Length
13	Day of year (= 1)	I*2
14	Latitude of subsatellite point (= 80 degrees)	I*2
⋮		
505	Day of year (= 1)	I*2
506	Latitude of subsatellite point (= -84 degrees)	I*2
507-510	Same as items 3 through 6	4 (R*4)
511	Day of year (= 16)	I*2
512-1020	Same as items 2 through 510	169 (I*2), 340 (R*4)
1021	Day of year (= 31)	I*2
1022-1530	Same as items 2 through 510	169 (I*2), 340 (R*4)
⋮		
11731	Day of year (= 346)	I*2
11732	Latitude of subsatellite point (= 84 degrees)	I*2
11733-12240	Same as items 3 through 510	168 (I*2), 340 (R*4)

SECTION 6 - EXISTING SOFTWARE

Many subroutines exist which perform functions identical to (or similar to) some of the functions described in Section 5. Tables 6-1 through 6-3 list these subroutines and indicate the subsection in Section 5 to which the subroutine is applicable. Libraries containing the subroutines are also included.

Table 6-1. Existing Subroutines Suitable for Use in the Telemetry Processing Functions

SUBROUTINE	APPLICABLE TO SECTION NO.	MODIFICATION REQUIRED	LIBRARY	REFERENCE
BT2HEX	5.2.3.2	NONE	ATTIT.GEOSC.PACKED.FORT	7
CKQLTY	5.2.3.5	MINOR	ATTIT.GEOSC.PACKED.FORT	7
CONVRT	5.2.3.4	MINOR	ATTIT.GEOSC.PACKED.FORT	7
DSCCHK	5.2.3.4	NONE	ATTIT.GEOSC.PACKED.FORT	7
FINOUT	5.2.3.7	MODERATE	ATTIT.GEOSC.PACKED.FORT	7
GMT CHK	5.2.3.6	NONE	ATTIT.GEOSC.PACKED.FORT	7
RDUNPK	5.2.3.2, 5.2.3.3	MODERATE	ATTIT.GEOSC.PACKED.FORT	7
READER	5.2.3.2	MINOR	ATTIT.GEOSC.PACKED.FORT	7
SADCON		MINOR	ATTIT.GEOSC.PACKED.FORT	7
SEARCH	5.2.3.2	MINOR	ATTIT.GEOSC.PACKED.FORT	7
STABIT	5.2.3.4	MINOR	ATTIT.GEOSC.PACKED.FORT	7
STATON	5.2.3.3	NONE	ATTIT.GEOSC.PACKED.FORT	7
TIMCHK	5.2.3.6	NONE	ATTIT.GEOSC.PACKED.FORT	7
TMDRIV	5.2.3	MODERATE	ATTIT.GEOSC.PACKED.FORT	7
UNPACK	5.2.3.3	MODERATE	ATTIT.GEOSC.PACKED.FORT	7
TCGCHK	5.2.3.6	NONE	ATTIT.GEOSC.PACKED.FORT	7
NAMLST	5.2.3.1	MODERATE	ATTIT.CTS.PACKED.FSD.FORT	NONE

Table 6-2. Existing Subroutines Suitable for Use in the Data Adjustment Function

SUBROUTINE	APPLICABLE TO SECTION NO.	MODIFICATION REQUIRED	LIBRARY	REFERENCE
CONES8	5.3.3.2	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
EPHEMX	5.3.3.1	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
ORBGEN	5.3.3.1	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
MAGFLD	5.3.3.1	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
SUN1X	5.3.3.1	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
GETHDR	5.3.3.1	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
RO1TAP	5.3.3.1	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
SUNRD	5.3.3.1	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
UNVEC	5.3.3.2 THROUGH 5.3.3.5	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
VEC	5.3.3.2 THROUGH 5.3.3.5	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
ANGLED	5.3.3.2 THROUGH 5.3.3.5	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
TCON20	5.3.3	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
TCON40	5.3.3	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
CNVERT	5.3.3.3	MINOR	ATTIT.GEOSC.PACKED.FORT	7
FITEDT	5.3.3.5	MINOR	ATTIT.GEOSC.PACKED.FORT	7
RMAT	5.3.3.3	NONE	ATTIT.GEOSC.PACKED.FORT	7
SUNBDY	5.3.3.3	MINOR	ATTIT.GEOSC.PACKED.FORT	7
SUNMAT	5.3.3.3	MINOR	ATTIT.GEOSC.PACKED.FORT	7
POLYFT	5.3.3.4	NONE	ATTIT.GEOSC.PACKED.FORT	7

Table 6-3. Existing Subroutines Suitable for Use in the Attitude Determination Function

SUBROUTINE	APPLICABLE TO SECTION NO.	MODIFICATION REQUIRED	LIBRARY	REFERENCE
DC	5.4.3.4	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
UNVEC	5.4.3.2 THROUGH 5.4.3.5	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
VEC	5.4.3.2 THROUGH 5.4.3.5	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
ANGLED	5.4.3.2 THROUGH 5.4.3.5	NONE	ATTIT.ATTMAIN.UTIL.PACK.FORT	8
DRVCNP	5.4.3.5	NONE	ATTIT.GEOSC.PACKED.FORT	7
TPPPRY	5.4.3.3	MINOR	ATTIT.GEOSC.PACKED.FORT	7
ATTFIT	5.4.3.5	MINOR	ATTIT.GEOSC.PACKED.FORT	7
DISATT	5.4.3.3	MODERATE	ATTIT.GEOSC.PACKED.FORT	7

SECTION 7 - AEM-A/HCMM SIMULATOR SPECIFICATIONS

This section describes the specifications for the HCMM dynamics and data simulator, which will satisfy the requirements outlined in Section 2.

7.1 SIMULATOR REQUIREMENTS AND CAPABILITIES

The HCMM simulator will provide data similar in form and content to that expected from the HCMM spacecraft to thoroughly test and evaluate all components of the HCMM ground support software, to train HCMM ground support attitude determination and control personnel, and to evaluate the performance and limitations of the onboard control system.

To meet these requirements, the simulator will operate in batch mode. Spacecraft dynamics, including all onboard control system modes and disturbance torques, will be simulated. The output data sets are shown in Figure 7-1 and described in Section 7.6 These include the following:

- Simulated MSOCC and IPD telemetry data
- Simulated control system data (wheel and electromagnet) for the residual dipole bias determination utility
- Simulated infrared scanner and Sun sensor data for the scanner bias determination utility
- Analytical data, including a truth model, to test and evaluate algorithms and to exercise the attitude determination system during development and acceptance testing
- Attitude history file for the residual dipole bias determination utility

The simulator will provide the following data types:

- Sun sensor angles and selected sensor head identification (ID)
- Magnetometer magnetic field and magnetic field rate

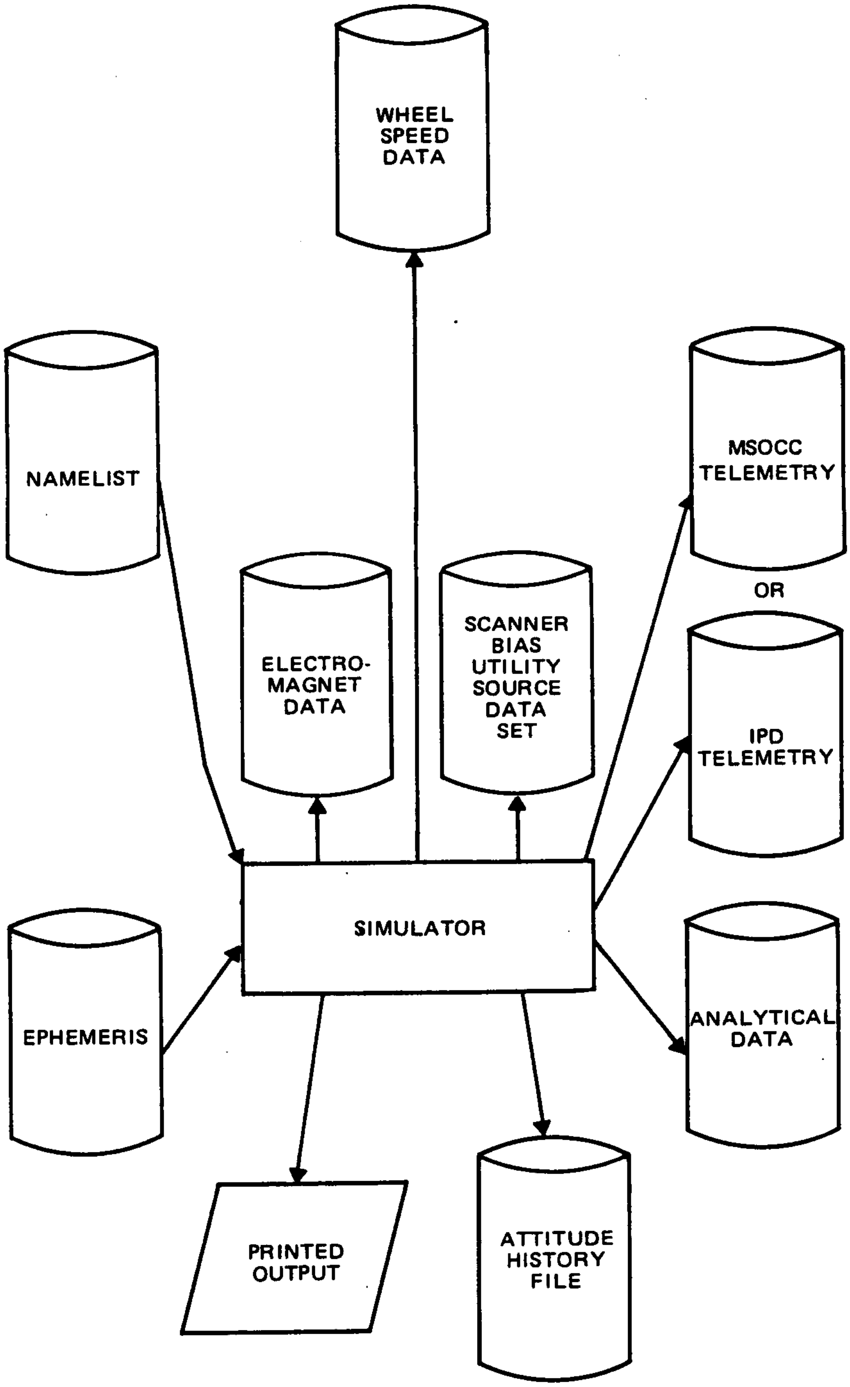


Figure 7-1. HCMM Simulator Data Flow

- Infrared scanner duty cycles, pitch error, roll error, and scan-wheel speed
- Electromagnet dipoles
- Control system status flags, including the roll bias command
- Attached Greenwich mean time (GMT)
- Spacecraft clock

Each data type will be simulated with sufficient fidelity to adequately test the ground support models. The sensor modèl fidelity of the simulator will, wherever possible, exceed that of the attitude determination system. Random sensor noise, systematic sensor bias, and random and systematic telemetry bit errors will be simulated.

Figure 7-2 illustrates the functional flow of the simulator. The initialization functions include reading the system NAMELIST, performing various initialization functions for the integrator, converting the input attitude state from various convenient formats (attitude of wheel axis; pitch, roll, and yaw; yaw axis along velocity vector) to Euler parameters, and directing the simulator flow. The equations of motion are then integrated using an Adams-Bashforth integrator. The Adams-Bashforth integrator is a variable step size predictor-corrector and is preferred over other methods in current use, such as the Runge-Kutta and Adams-Moulton, for the following two basic reasons:

1. It is more efficient and therefore faster because it does not require matrix inversion (the Adams-Moulton variable step integrator inverts two 5-by-5 matrices each time step).
2. It can handle impulsive torques such as those induced when the onboard control system switches the polarity of the electromagnets.

Ephemeris and a magnetic field model are required to compute environmental torques and to simulate sensor data. The accuracy requirements of Sun,

INITIALIZATION

- READ NAMELIST
- INITIALIZE SIMULATOR
- CONVERT INPUT STATE TO EULER PARAMETERS AND BODY RATES

INTEGRATE EQUATIONS OF MOTION

- TORQUE MODELS
  - GRAVITY GRADIENT
  - MAGNETIC DIPOLE
  - WHEEL REACTION
- EPHEMERIS
- MAGNETIC FIELD MODEL

SIMULATE SENSOR OUTPUT

- SUN SENSORS
- INFRARED SCANNER
- MAGNETOMETERS

SIMULATE CONTROL SYSTEM OUTPUT

- ELECTROMAGNETS

FORMAT OUTPUT DATA

- ADD NOISE
- ADD SYSTEMATIC ERRORS

Figure 7-2. HCMM Simulator Functional Flow spacecraft, and magnetic field ephemeris are identical to those specified in Section 4.2. Spacecraft position and velocity ephemeris is obtained from either an internal orbit generator or a data set in EPHEM format. The internal orbit generator used must be the same as that used in the Attitude Determination System (ADS) for system testing. The simulated truth model attitude and ephemeris is then used to model the output of the magnetometers, Sun sensors, and infrared scanner. The infrared scanner and magnetometer data is passed to the control system, which computes commands to the electromagnets and scanwheel. Finally, the sensor and control system output is contaminated with noise and systematic errors, reformatted, and written to the various data sets shown in Figure 7-1.

7.2 OPERATIONAL MODES

The simulator will be operated in a batch mode to exercise the ADS. All possible control modes--acquisition, on-orbit, and orbit adjust--will be provided. The simulator will be used to provide data to test the definitive mode of the ADS and to evaluate the performance of various utility programs and the onboard control system. Provision for generating data only over specified telemetry stations will be provided.

Truth model printer plots will provide visibility for interactive operation. Displays of attitude and attitude rate, shown in Figures 7-3 to 7-5, are required.

7.3 DYNAMICS

This subsection describes the equations of motion for the AEM-A/HCMM spacecraft. An Adams-Bashforth algorithm to integrate the equations of motion is outlined.

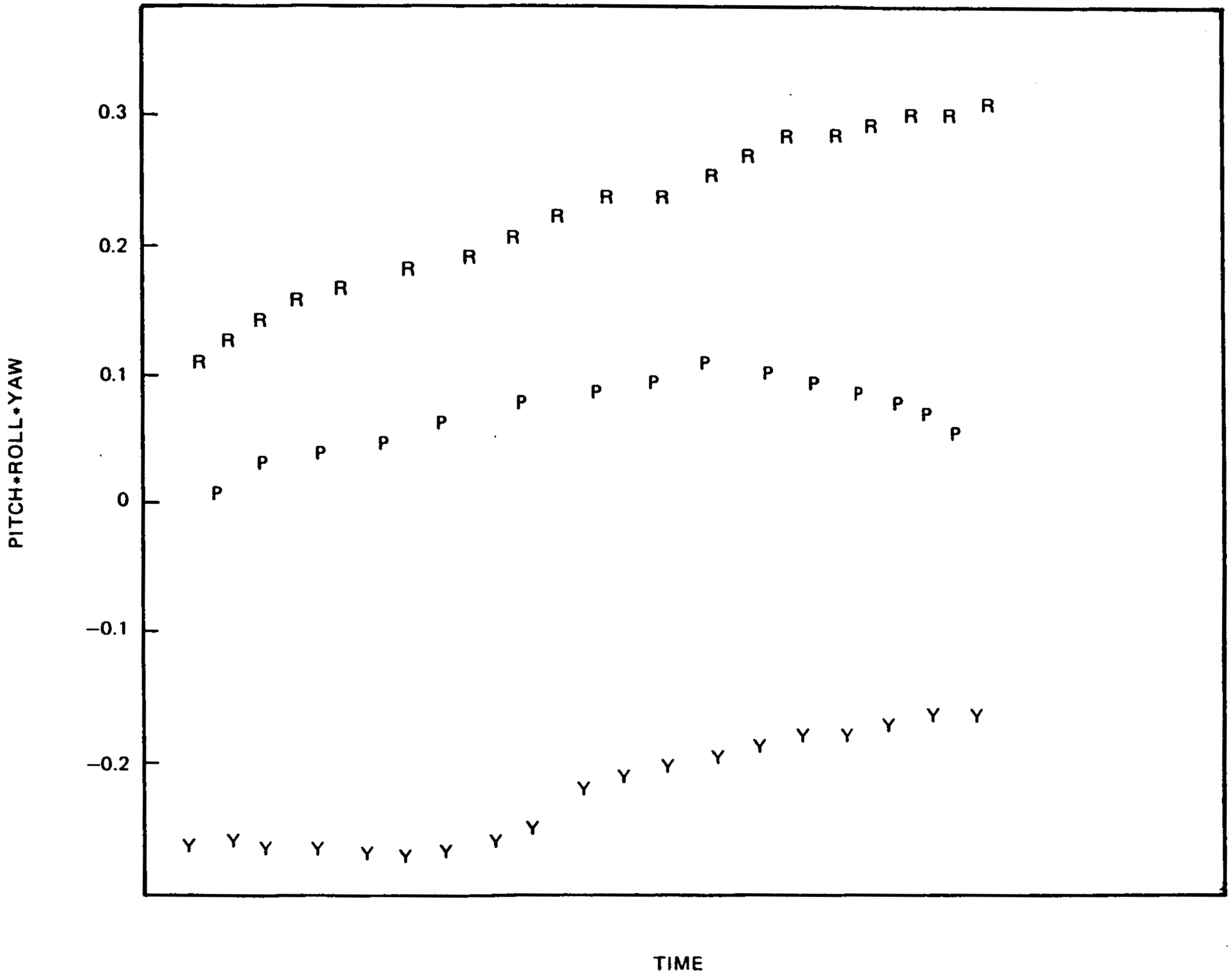


Figure 7–3. Plot of Pitch, Roll, and Yaw Angles

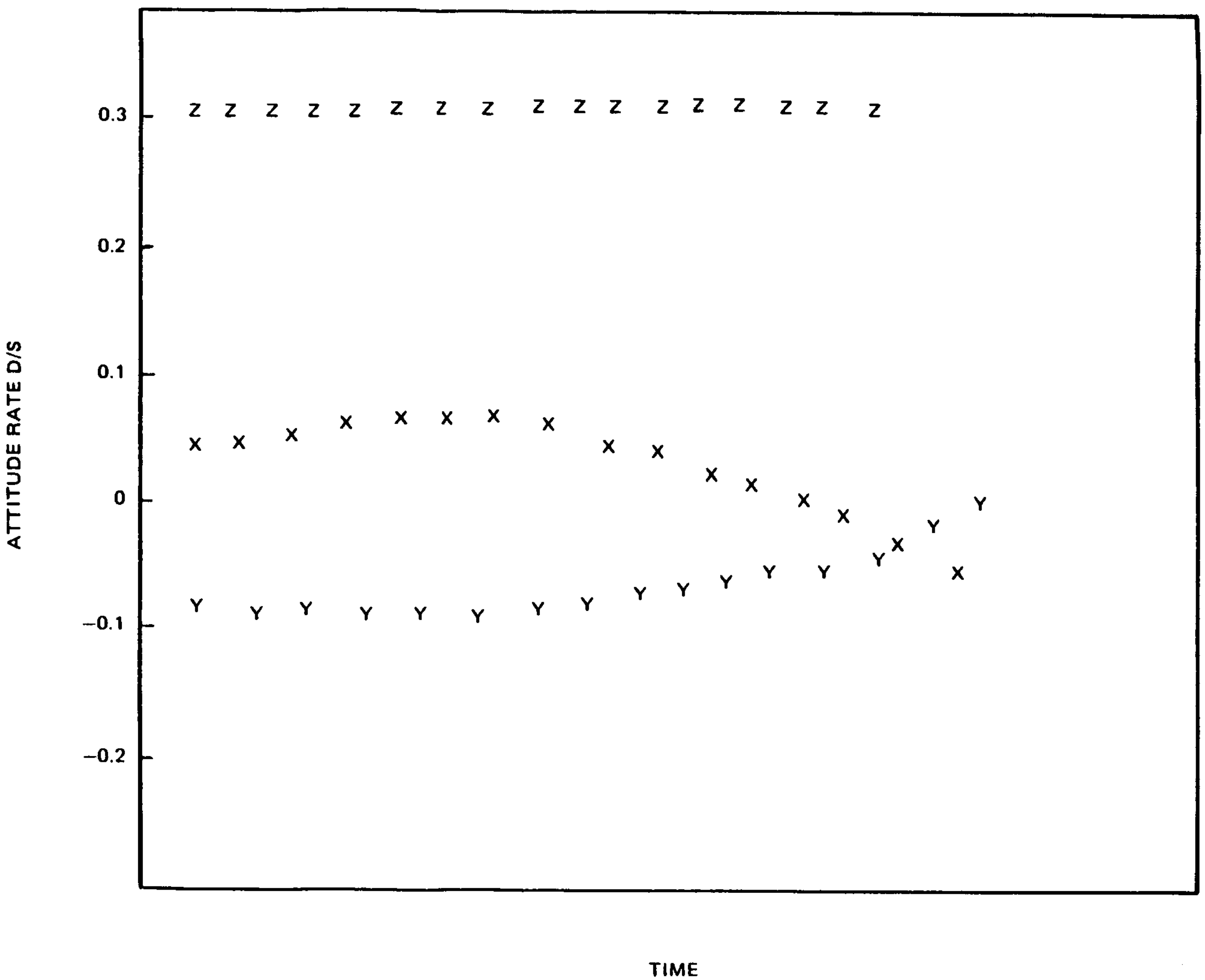


Figure 7-4. Plot of Body Rates, ω_x , ω_y , ω_z

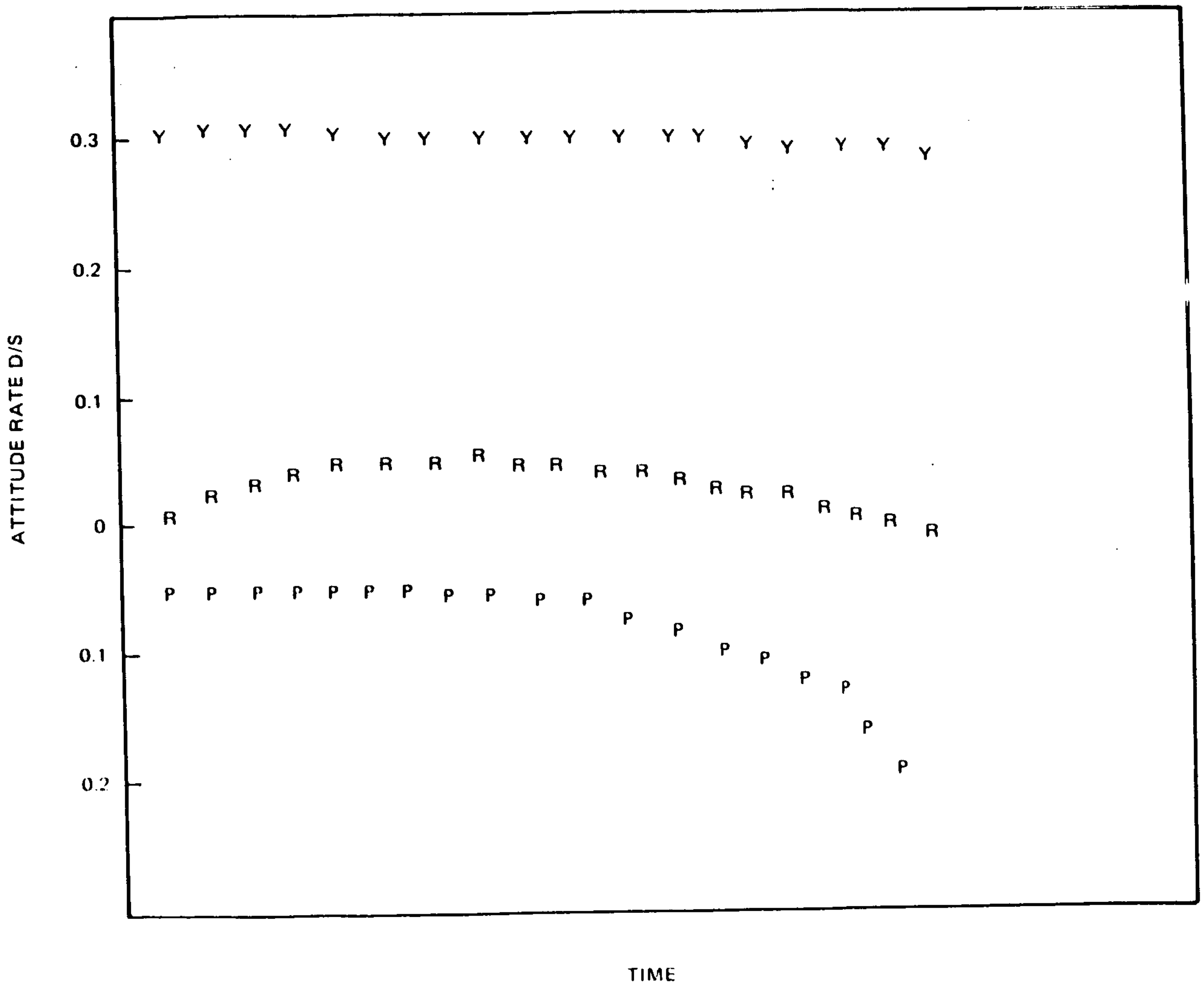


Figure 7-5. Plot of Pitch, Roll, and Yaw Rates

7.3.1 Equations of Motion

The attitude dynamics of the HCMM spacecraft can be described by Euler's equations of motion for a rigid body rotating under the influence of external torques. The equations are (in the body frame) as follows:

$$I_1\dot{\omega}_1 = (I_2 - I_3)\ \omega_2\omega_3 + \tau_1$$
$$I_2\dot{\omega}_2 = (I_3 - I_1)\ \omega_3\omega_1 + \tau_2 \qquad (7\text{-}1)$$
$$I_3\dot{\omega}_3 = (I_1 - I_2)\ \omega_1\omega_2 + \tau_3$$

where I_1, I_2, and I_3 are the principal moments of inertia, $\bar{\omega}$ is the satellite angular velocity vector, $\dot{\omega}_i$ is the time derivative of the ith component of the angular velocity vector, and $\bar{\tau}$ is the sum of all external torques. These equations are convenient for obtaining the components of the angular velocity with respect to the body axes.

To obtain the orientation of the body axes with respect to an inertial reference frame, additional parameters are needed. Let [A] be the rotation matrix which transforms vectors from inertial to body coordinates, i.e., $[A]\bar{v}_I = \bar{v}_B$. The A matrix can be written in terms of the 3-1-3 (ϕ, θ, ψ) Euler angles or in terms of the Euler symmetric parameters (ζ_1, ζ_2, ζ_3, ζ_4). The Euler parameters are defined in terms of the Euler angles by the following relations:

$$\zeta_1 = \sin\frac{\theta}{2}\cos\left(\frac{\phi - \psi}{2}\right)$$
$$\zeta_2 = \sin\frac{\theta}{2}\sin\left(\frac{\phi - \psi}{2}\right)$$
$$\zeta_3 = \cos\frac{\theta}{2}\sin\left(\frac{\phi + \psi}{2}\right) \qquad (7\text{-}2)$$
$$\zeta_4 = \cos\frac{\theta}{2}\cos\left(\frac{\phi + \psi}{2}\right)$$

Differentiating these equations with respect to time and using the substitution (see Reference 9)

$$\omega_1 = \dot{\phi} \sin \theta \sin \psi + \dot{\theta} \cos \psi$$

$$\omega_2 = \dot{\phi} \sin \theta \cos \psi - \dot{\theta} \sin \psi \qquad (7\text{-}3)$$

$$\omega_3 = \dot{\phi} \cos \theta + \dot{\psi}$$

yields the following result:

$$\dot{\zeta}_1 = \frac{1}{2} (\omega_3 \zeta_2 - \omega_2 \zeta_3 + \omega_1 \zeta_4)$$

$$\dot{\zeta}_2 = \frac{1}{2} (-\omega_3 \zeta_1 + \omega_1 \zeta_3 + \omega_2 \zeta_4)$$

$$\dot{\zeta}_3 = \frac{1}{2} (\omega_2 \zeta_1 - \omega_1 \zeta_2 + \omega_3 \zeta_4) \qquad (7\text{-}4)$$

$$\dot{\zeta}_4 = -\frac{1}{2} (\omega_1 \zeta_1 + \omega_2 \zeta_2 + \omega_3 \zeta_3)$$

Equations (7-1) and (7-4) describe the dynamic behavior of the satellite under the influence of external torques. Note that the norm of the Euler parameters is unity.

The torques acting on the satellite include disturbance torques, gravity-gradient and solar radiation pressure, and the magnetic and wheel control torques.

The gravitational torque experienced by the satellite in a field μ/E is given by

$$\bar{\tau}_{gg} = \frac{3\mu}{E^3} (\hat{E} \times (\overleftrightarrow{I} \cdot \hat{E})) \qquad (7\text{-}5)$$

where $\overleftrightarrow{I}$ is the moment of inertia tensor and $\bar{E}$ is radius vector from the center of the Earth to the center of mass of the satellite. Assuming that $\overleftrightarrow{I}$ is diagonal, the torque in the body is

$$\bar{\tau}_{gg} = -3\Omega_0^2 \begin{bmatrix} K_{23} & K_{33} & (I_2 - I_3) \\ K_{33} & K_{13} & (I_3 - I_1) \\ K_{13} & K_{23} & (I_1 - I_2) \end{bmatrix} \tag{7-6}$$

where $\Omega_0 \approx 0.063$ degree per second is the orbital rate, and [K] is the orbital to body coordinate transformation matrix. Expressed in terms of the orbit normal, $\hat{n}$, and radius vector, $\hat{E}$, the K matrix is

$$[K] = [A] [\hat{n} \times \hat{E} \;\vert\; -\hat{n} \;\vert\; -\hat{E}] \equiv [A] [R]^T \tag{7-7}$$

The momentum wheel torque is

$$\bar{\tau}_{\omega} = \bar{L} \times \bar{\omega} + \bar{\tau}_B \tag{7-8}$$

where $\bar{L}$ is the angular momentum of the wheel (4.06 ± 0.08 slug-feet^2 per second in the $-\hat{j}$ direction is nominal), $\bar{\omega}$ is the angular velocity of the satellite (radians per second), and $\bar{\tau}_B$ is the bearing torque (slug-feet^2 per second^2). At steady state $\bar{\tau}_B = 0$; during acceleration, $\bar{\tau}_B$ is along $-\hat{j}$, and during deceleration, $\bar{\tau}_B$ is along $+\hat{j}$. A typical bearing torque is 0.018 slug-feet^2 per second^2. The wheel moment of inertia in 0.02 slug-feet^2 per second^2 and nominal wheel speed is 1940 rpm with a range of 0 to 2200 rpm.

The solar radiation torque is small for the HCMM orbit; it is adequate to assume a zero value in the body,

$$\bar{\tau}_{SRP} = (0, 0, 0)^T \tag{7-9}$$

The magnetic torque (in slug-feet^2 per second^2) results from the interaction of the spacecraft's magnetic dipole with the Earth's magnetic field,

$$\bar{\tau}_M = K_M \, \bar{D} \times \bar{M} \tag{7-10}$$

where $\bar{D} = \bar{D}_C + \bar{D}_R + \bar{D}_{C0}$ is the sum of the commanded, residual, and null magnetic dipole (pole-centimeters); $\bar{M}$ is the ambient magnetic field (millioersteds); and $K_M = 7.3756 \times 10^{-11}$ is a constant. The commanded magnetic dipole is computed from the control mode, magnetometer, and IR scanner data, and scanner wheel speed.

The acquisition mode control law is

$$\bar{D}_C = -K_{SAT} \text{ sign } (\dot{\bar{M}}') \tag{7-11}$$

where $\bar{M}' = (M'_x, M'_y, M'_z)$ is the magnetometer data used by the control system, defined in Equation (7-38), K_{SAT} = 10,000 pole-centimeters, and

$$\text{sign } (x) = \begin{Bmatrix} 1 \\ -1 \end{Bmatrix} \text{ for } \begin{Bmatrix} x \geq 0 \\ x < 0 \end{Bmatrix} \tag{7-12}$$

The on-orbit control law consists of precession control, nutation control, and momentum unloading. The precession/nutation control law drives the Y-axis dipole,

$$D_{Cy} = K_1 \, M'_x \, \Delta\varphi + K_2 \, \dot{M}'_y \tag{7-13}$$

where $\Delta\varphi$ is the roll error in volts, and $\bar{D}_C = (D_{Cx}, D_{Cy}, D_{Cz})^T$ is the commanded dipole.

The momentum unloading control law drives the X and Z axis dipoles,

$$\begin{aligned} D_{Cx} &= -K_a \text{ sign } (M'_x \Delta S) \\ D_{Cz} &= K_a \text{ sign } (M'_z \Delta S) \end{aligned} \qquad (7\text{-}14)$$

where $\Delta S = S - S_W$

S = wheel speed (rpm)

S_W = nominal wheel speed (rpm)

K_a is the gain such that $K_a = 0$ if $|\Delta S| < \Delta S_{MAX}$, but when $|\Delta S| \geq \Delta S_{MAX}$, K_a is set to K_{USAT} (saturation gain) until $|\Delta S| < \Delta S_{TOL}$.

The nominal values are

$$\begin{aligned} K_1 &= -500 \text{ pole-centimeters per volt} \\ K_2 &= -6 \times 10^4 \text{ pole-centimeter-seconds per millioersted} \\ K_{USAT} &= 10{,}000 \text{ pole-centimeters} \\ S_W &= 1940 \text{ rpm} \\ \Delta S_{MAX} &= 40 \text{ rpm} \\ \Delta S_{TOL} &= 2 \text{ rpm} \end{aligned} \qquad (7\text{-}15)$$

Near the poles, precession and nutation control is disabled (blanking), i.e., $K_1 = K_2 = 0$, when $|M'_z/M'_x| > f_M$, where the nominal value is $f_M = 1.4$.

Pitch control is achieved by adjusting the wheel voltage or torque,

$$\bar{\tau}_B = +\tau_B \hat{j} \tag{7-16}$$

where $\tau'_B = K_\theta \, \Delta\theta + K_{\dot{\theta}} \, \Delta\dot{\theta}$ and $\tau_B = \text{sign}\,(\tau'_B) \min\,(|\tau'_B|, \tau_{MAX})$

$$\begin{aligned}
\Delta\theta &= \text{pitch angle error (volts)} \\
\Delta\dot{\theta} &= \text{pitch angle error rate (volts per second)} \\
K_\theta &\approx -2.8 \times 10^{-3} \text{ (foot-pounds per volt)} \\
K_{\dot{\theta}} &\approx -1.9 \times 10^{-1} \text{ (foot-pound-seconds per volt)} \\
\tau_{MAX} &= 1.4 \times 10^{-1} \text{ (foot-pounds)}
\end{aligned} \tag{7-17}$$

Because the wheel momentum is along the negative Y-body axis, a positive pitch error is nulled by decreasing the wheel speed, or torquing the wheel in the +Y direction.

The orbit adjust burn will cause an impulsive disturbance torque during the burn internal $t_{OA} < t < t_{OA} + \Delta t_{OA}$

$$\bar{\tau}_T = \Delta\bar{r} \times \bar{F} \, \{g(t - t_{OA}) - g(t - t_{OA} - \Delta t_{OA})\} \tag{7-18}$$

where the step function $g(x)$ is

$$g(x) = \begin{Bmatrix} 0 \\ 1 \end{Bmatrix} \text{ for } \begin{Bmatrix} x < 0 \\ x \geq 0 \end{Bmatrix} \tag{7-19}$$

and where $\Delta\bar{r} \times \bar{F}$ is the orbit adjust torque.

7.3.2 Integrators

Several integrators have been used for modeling spacecraft dynamics, including the Runge-Kutta, Adams-Moulton, and Adams-Bashforth (Reference 10). The Adams-Bashforth integrator, which handles impulsive torques, has been implemented for the Solar Maximum Mission/Multimission Modular Spacecraft (SMM/MMS) dynamics simulator and appears well suited for the HCMM application. The equations of motion (equations (7-1) and (7-4)) may be written in the compact form,

$$\frac{d\bar{y}}{dt} = f(t, \bar{y}) \tag{7-20}$$

Dropping the vector notation for convenience, the predictor at the nth step uses the current and four back values of the function to predict the function at the n+1st step,

$$y_{n+1}^{(p)} = y_n + h\left[1 + \frac{1}{2}\nabla + \frac{5}{12}\nabla^2 + \frac{3}{8}\nabla^3 + \frac{251}{720}\nabla^4\right]f_n \tag{7-21}$$

where

$$h = \text{step size}$$

$$\begin{aligned}
\nabla f_n &= f_n - f_{n-1} \\
\nabla^2 f_n &= f_n - 2f_{n-1} + f_{n-2} \\
\nabla^3 f_n &= f_n - 3f_{n-1} + 3f_{n-2} - f_{n-3} \\
\nabla^4 f_n &= f_n - 4f_{n-1} + 6f_{n-2} - 4f_{n-3} + f_{n-4}
\end{aligned} \tag{7-22}$$

The derivative is then evaluated at the n+1st step, $f^{(p)}_{n+1} = f\left(t_{n+1}, y^{(p)}_{n+1}\right)$, and the corrector uses the current and three back values to correct the function,

$$y^{(c)}_{n+1} = y_n + \frac{h}{720}\left[469 + 109\,\nabla + 49\,\nabla^2 + 19\,\nabla^3\right] f_n + \frac{251h}{720} f_{n+1} \qquad (7\text{-}23)$$

The difference between $y^{(p)}$ and $y^{(c)}$ is monitored at each step, for each equation, and if $|y^{(p)} - y^{(c)}| > N_D\epsilon$, where ϵ is an input parameter, the step size is halved; if $|y^{(p)} - y^{(c)}| < \epsilon/N_D$, the step size is doubled. Control of the doubling is maintained so that the function is computed at a preselected interval for data output. The nominal value is $N_D = 5$.

7.3.3 Initialization

This section describes the transformations from reference systems convenient for input/output to the inertial system (Euler parameters) used by the integrator.

7.3.3.1 Orbital Coordinates

A convenient parameterization of the [K] matrix is the 2-1-3 Euler angles denoted pitch (p), roll (r), and yaw (y),

$$[K] = \begin{bmatrix} cpcy + spsrsy & crsy & -spcy + cpsrsy \\ -cpsy + spsrcy & crcy & spsy + cpsrcy \\ spcr & -sr & cpcr \end{bmatrix} \qquad (7\text{-}24)$$

where the shorthand notation c = cosine, s = sine has been used. The attitude matrix and angular velocity are

$$[A] = [K]\,[R] \qquad (7\text{-}25a)$$

$$\bar{\omega} = \begin{bmatrix} \text{sycr}\,(\dot{p} - \Omega_o) + \text{cy}\dot{r} \\ \text{cycr}\,(\dot{p} - \Omega_o) - \text{sy}\dot{r} \\ -\text{sr}\,(\dot{p} - \Omega_o) + \dot{y} \end{bmatrix} \qquad (7\text{-}25b)$$

where [R] is the inertial to orbital transformation matrix defined in Section 4.1 and where Ω_o is the orbital angular velocity.

The 3-1-3 (ϕ, θ, ψ) Euler angles which parameterize the [A] matrix are

$$\begin{aligned} \theta &= \cos^{-1}(A_{33}) \\ \phi &= \tan^{-1}(A_{31}/-A_{32}) \\ \psi &= \tan^{-1}(A_{13}/A_{23}) \end{aligned} \qquad (7\text{-}26)$$

and

$$[A]_{313} = \left[\begin{array}{c|c|c} c\psi c\phi - c\theta s\phi s\psi & c\psi s\phi + c\theta c\phi s\psi & s\psi s\theta \\ -s\psi c\phi - c\theta s\phi c\psi & -s\psi s\phi + c\theta c\phi c\psi & c\psi s\theta \\ s\theta s\phi & -s\theta c\phi & c\theta \end{array}\right] \qquad (7\text{-}27)$$

In the singular case when $A_{31} = A_{32} = 0$, the Euler angles are $\theta = \psi = 0$ and $\phi = \tan^{-1}(A_{21}/A_{11})$.

The inverse transformation is

$$\begin{aligned} \dot{p} &= (\omega_1\,\text{sy} + \omega_2\,\text{cy})\,/\,\text{cr} + \Omega_o \\ \dot{r} &= \omega_1\,\text{cy} - \omega_2\,\text{sy} \\ \dot{y} &= \omega_3 + (\omega_1\,\text{sy} + \omega_2\,\text{cy})\tan r \end{aligned} \qquad (7\text{-}28)$$

7.3.3.2 3-1-2 Inertial Coordinates

The spacecraft Y-axis may be fixed in inertial space via a 3-1-2 $(\theta_1, \theta_2, \theta_3)$ Euler transformation. The $[A]_{312}$ matrix is

$$[A]_{312} = \begin{bmatrix} c\theta_3 c\theta_1 - s\theta_3 s\theta_2 s\theta_1 & c\theta_3 s\theta_1 + s\theta_3 s\theta_2 c\theta_1 & -s\theta_3 c\theta_2 \\ -c\theta_2 s\theta_1 & c\theta_2 c\theta_1 & s\theta_2 \\ s\theta_3 c\theta_1 + c\theta_3 s\theta_2 s\theta_1 & s\theta_3 s\theta_1 - c\theta_3 s\theta_2 c\theta_1 & c\theta_3 c\theta_2 \end{bmatrix} \quad (7\text{-}29)$$

The right ascension and declination of the Y-axis are

$$\begin{aligned} \alpha_y &= \theta_1 - 90^\circ \\ \delta_y &= 90^\circ - \theta_2 \end{aligned} \quad (7\text{-}30)$$

7.3.3.3 Velocity Vector Attitude

The Z-axis of the spacecraft is aligned along the velocity vector at separation. This is equivalent to a 3-1-3 inertial attitude with

$$\phi = \tan^{-1}(v_2/v_1) + 90^\circ$$

$$\theta = 90^\circ - \sin^{-1}(v_3/|v|)$$

$$\psi = \text{indeterminate}$$

where $\bar{v} = (v_1, v_2, v_3)^T$ is the spacecraft velocity vector.

7.4 ATTITUDE DETERMINATION AND CONTROL HARDWARE MODELING

This subsection describes the algorithms required to model the HCMM attitude determination and control hardware. Attitude sensor outputs modeled include the Adcole Sun sensors, Schoenstedt fluxgate magnetometers, and an Ithaco infrared scanwheel.[1] Control hardware output includes the electromagnet dipoles, magnetic field rates, and scanwheel speed.

7.4.1 Magnetometers

The net magnetic field in the spacecraft is

$$\bar{M} = [A]\, \bar{M}_I + \bar{b}_B \tag{7-31}$$

where $\bar{M}_I$ is the inertial field obtained from a field model, $\bar{b}_B$ is a bias field, and $[A]$ is the 3-by-3 attitude matrix defined in Section 7.3.1.

If the orientation of the ith magnetometer is defined by the cone and azimuth angles θ_i and ϕ_i, the alignment matrix $[M]$ is

$$[M] = [P] \begin{bmatrix} \sin\theta_x \cos\varphi_x & \sin\theta_x \sin\varphi_x & \cos\theta_x \\ \sin\theta_y \cos\varphi_y & \sin\theta_y \sin\varphi_y & \cos\theta_y \\ \sin\theta_z \cos\varphi_z & \sin\theta_z \sin\varphi_z & \cos\theta_z \end{bmatrix} \tag{7-32}$$

where the nominal alignment angles are given in Table 7-1 and $[P]$ is a diagonal matrix whose elements are the magnetometer scale factors.

[1]Scanwheel is a registered trademark of the Ithaco Corporation.

Table 7-1. Magnetometer Angles

MAGNETOMETER	θ (DEGREES)	ϕ (DEGREES)
x	90	0
y	90	90
z	0	INDETERMINATE

The observed magnetometer voltage is

$$\bar{V} = [M]\left([A]\,\bar{M}_I + \bar{b}_B\right) + \bar{V}_0 \tag{7-33}$$

where $\bar{V}_0$ is the zero offset voltage and is nominally a null vector. Finally, the analog to digital conversion (ADC) is

$$\bar{N}_V = \begin{bmatrix} \text{Int}\,(C_x V_x + 0.5) \\ \text{Int}\,(C_y V_y + 0.5) \\ \text{Int}\,(C_z V_z + 0.5) \end{bmatrix} \tag{7-34}$$

where $\bar{V} = (V_x, V_y, V_z)^T$; C_x , C_y , and C_z are ADC factors; and the function Int(x) denotes the integral part of x . Assuming that ±600 millioersteds yield a full scale magnetometer voltage of ±10 volts and the ADC is 8 bits, the nominal values are

$$P_{11} = P_{22} = P_{33} = 0.01667 \text{ volts per millioersted}$$

$$P_{12} = P_{21} = P_{13} = P_{31} = P_{23} = P_{32} = 0 \tag{7-35}$$

$$C_x = C_y = C_z = 0.07843 \text{ counts per volt}$$

The bias field, $\bar{b}_B$, is composed of a constant term, $\bar{b}_{B0}$, plus a term induced by the electromagnets. The induced bias, in millioersteds (Reference 11), is

$$\bar{b}_{BM} = \frac{1000\ m}{r_{EM}^3} (3 \cos \theta\, \hat{r}_{EM} - \hat{n}) \tag{7-36}$$

where $\cos \theta = \hat{n} \cdot \hat{r}_{EM}$

$\bar{m} = m\hat{n}$ = vector sum of electromagnet dipoles, $\bar{D}_C + \bar{D}_{C0}$, (pole-centimeters)

$\bar{r}_{EM}$ = position vector from electromagnets to magnetometers (centimeters)

and the total magnetometer bias is

$$\bar{b}_B = \bar{b}_{B0} + \bar{b}_{BM} \tag{7-37}$$

Equation (7-36) assumes that the physical dimensions of the magnetometers and electromagnets are small compared to the distance between them. The magnetometer data used by the control system is decoupled from the electromagnets by

$$\bar{M}' = \bar{M} - [C_D] (D_{Cx}, D_{Cy}, D_{Cz})^T \tag{7-38}$$

where $[C_D]$ is a matrix which defines the orientation of the magnetometers relative to the electromagnets and $\bar{D}_C$ is the commanded dipole.

This compensation for the effect of the electromagnets and the operation of the control laws (Equations (7-11), (7-13), and (7-14)) imply continuous knowledge of both the commanded dipole and the magnetometer data used by the control system. In general, $\bar{M}'$ and $\bar{D}_C$ cannot be solved simultaneously from the magnetometer compensation equation and the equation expressing the control law. For purposes of simulation, this interaction is modeled at the ith integration step in the following way:

1. Determine the attitude matrix, $[A]_i$ (Equations (7-1) and (7-4)), based on the torques evaluated at the previous time step, $\bar{\tau}_{i-1}$.

2. Calculate the net magnetic field in the spacecraft $\bar{M}_i$ (Equation (7-31)), based on the current attitude, $[A]_i$, the current inertial field, $\bar{M}_{Ii}$, the constant bias, $\bar{b}_{B0}$, and the previous value of the commanded dipole, $\bar{D}_{Ci-1}$, which determines the induced bias, $\bar{b}_{BM}$ (Equation (7-36)).

3. Calculate the magnetometer data used by the control system, $\bar{M}_i'$ (Equation (7-38)), based on the current magnetic field in the spacecraft, $\bar{M}_i$, and the previous value of the commanded dipole, $\bar{D}_{Ci-1}$.

4. Calculate the commanded dipole, $\bar{D}_{Ci}$ (Equations (7-11), (7-13), or (7-14)), based on the current value of the control system magnetometer data, $\bar{M}_i'$, or its derivative, $\dot{\bar{M}}_i'$.

7.4.2 <u>Sun Sensors</u>

The spacecraft to Sun position vector in the spacecraft body is

$$\hat{S}_B = [A]\,\hat{S}_I \tag{7-39}$$

where [A] is the attitude matrix and $\hat{S}_I$ is the spacecraft to Sun position vector in inertial coordinates. The correction for parallax, due to the spacecraft orbit, is less than 0.003 degree and need not be applied.

Digital Sun data is available provided that the Sun is visible from the spacecraft, e.g.,

$$-\hat{E} \cdot \hat{S} > \cos(\rho + \epsilon) \tag{7-40}$$

where $-\hat{E}$ is the nadir vector,

$$\rho = \sin^{-1}(6378.14/|\bar{E}|) \tag{7-41}$$

is the angular radius of the Earth, and ϵ is a small correction for the finite size of the Sun. ($\epsilon \approx 0.5$ degree implies the Sun is completely visible at the horizon; $\epsilon \approx -0.5$ degree implies the Sun is barely visible).

First, the automatic threshold adjust (ATA) output for the ith sensor head is computed by rotating $\hat{S}_B$ into each Sun sensor frame,

$$\hat{S}_S^i = [S_i]^T \hat{S}_B \tag{7-42}$$

The ATA output is

$$V_{ATA}^i = f_i Z_S^i \tag{7-43}$$

where $\hat{S}_S^i = \left(X_S^i, Y_S^i, Z_S^i\right)^T$ and f_i is a scale factor.

The telemetry output consists of the ID and angles for the sensor head with the largest ATA output, provided that the output exceeds a selected threshold.

The scale factor, f_i , and threshold are nominally unity and the cosine of 64 degrees, respectively, for each head. (In actual operation, the scale factors may differ between heads and are subject to hysteresis.)

The sensor output is

$$NA = \mathrm{Int}\left((a + a_0)/k_a + 0.5\right)$$
$$NB = \mathrm{Int}\left((b + b_0)/k_b + 0.5\right) \tag{7-44}$$

where

$$a = - X_S \gamma^{1/2} \tag{7-45a}$$

$$b = Y_S \gamma^{1/2} \tag{7-45b}$$

$$\gamma = h^2/\left(n^2 - X_S^2 - Y_S^2\right) \tag{7-45c}$$

$$a_0 = b_0 = \text{sensor null position} \approx 0.350625 \tag{7-45d}$$

$$k_a = k_b = \text{sensor resolution} \approx 0.00275 \tag{7-45e}$$

$$n = \text{sensor index of refraction} \approx 1.4553 \tag{7-45f}$$

$$h = \text{sensor slab thickness} \approx 0.448 \tag{7-45g}$$

The quantity γ cannot be negative provided $n \geq 1$. Note that both NA and NB must be less than 255; otherwise, the sensor ID is zero.

The sensor to body transformation matrix, $[S_i]$, is a Euler matrix, where the angles θ and ϕ define the sensor boresight in the body and ψ is a rotation about the boresight as shown in Figure 7-6.

$$[S] = \begin{bmatrix} -s\phi c\psi - c\theta c\phi s\psi & -c\theta c\phi c\psi + s\phi s\psi & s\theta c\phi \\ c\phi c\psi - c\theta s\phi s\psi & -c\theta s\phi c\psi - c\phi s\psi & s\theta s\phi \\ s\theta s\psi & s\theta c\psi & c\theta \end{bmatrix} \qquad (7\text{-}46)$$

Note that for $\psi = 0$, the sensor slit A is parallel to the spacecraft X-Y plane and the spacecraft Z-axis is in the sensor Y_S-Z_S plane.

The nominal alignment angles are given in Table 7-1 and the subscript i , in Equation (7-46), has been suppressed for convenience.

Table 7-2. Sun Sensor Alignment Angles in Degrees

SENSOR HEAD	ϕ_i (CLOCK)	θ_i (CONE)	ψ_i (PHASE)
1	− 90	50	45
2	− 60	137	−90
3	−120	137	−90

Figures 7-7 and 7-8 define the Sun sensor reference frame. Telemetered sensor output consists of 19 bits: a 3-bit sensor head ID plus two 8-bit, Gray-encoded angles.

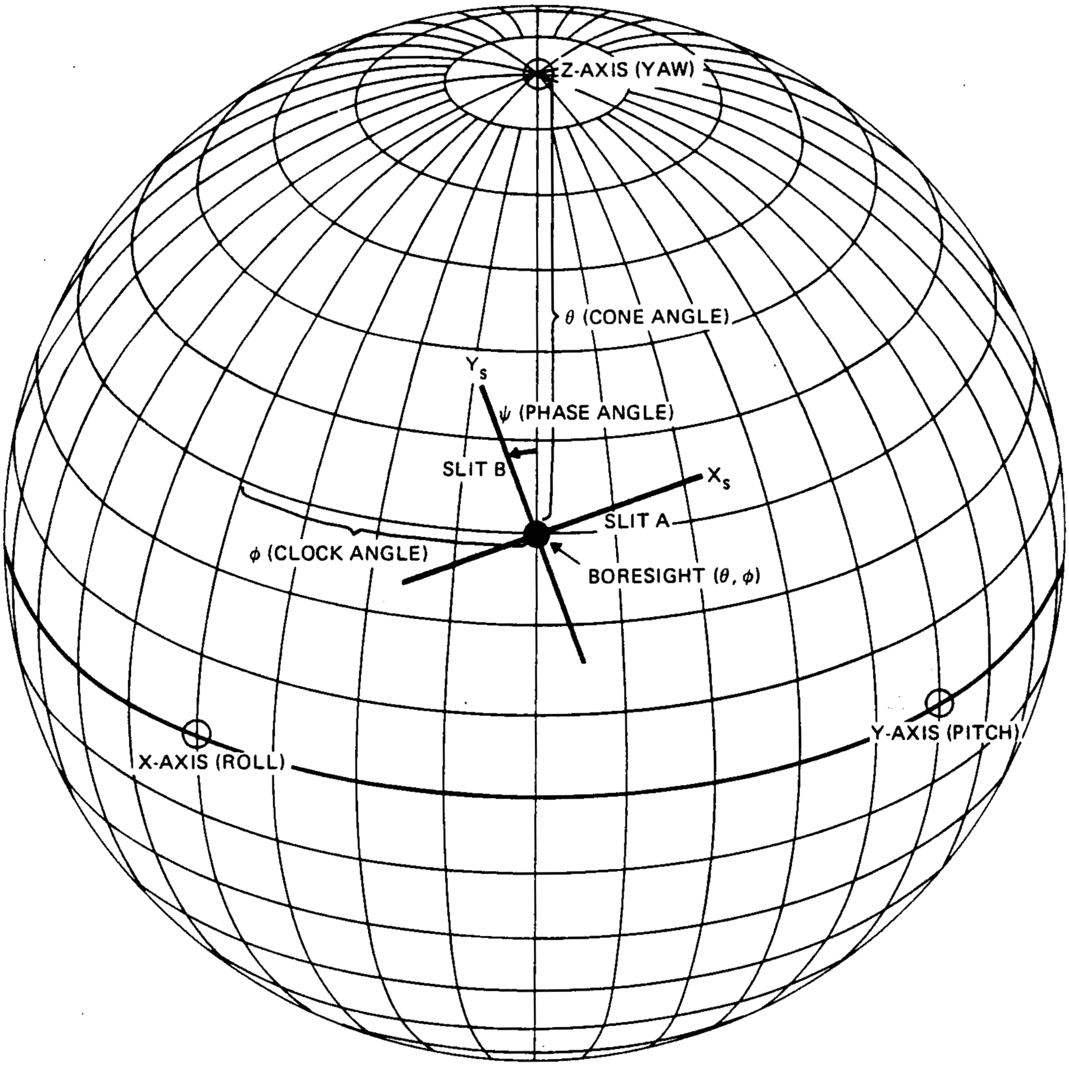


Figure 7-6. Orientation of Two-Axis Sun Sensors in Spacecraft Body

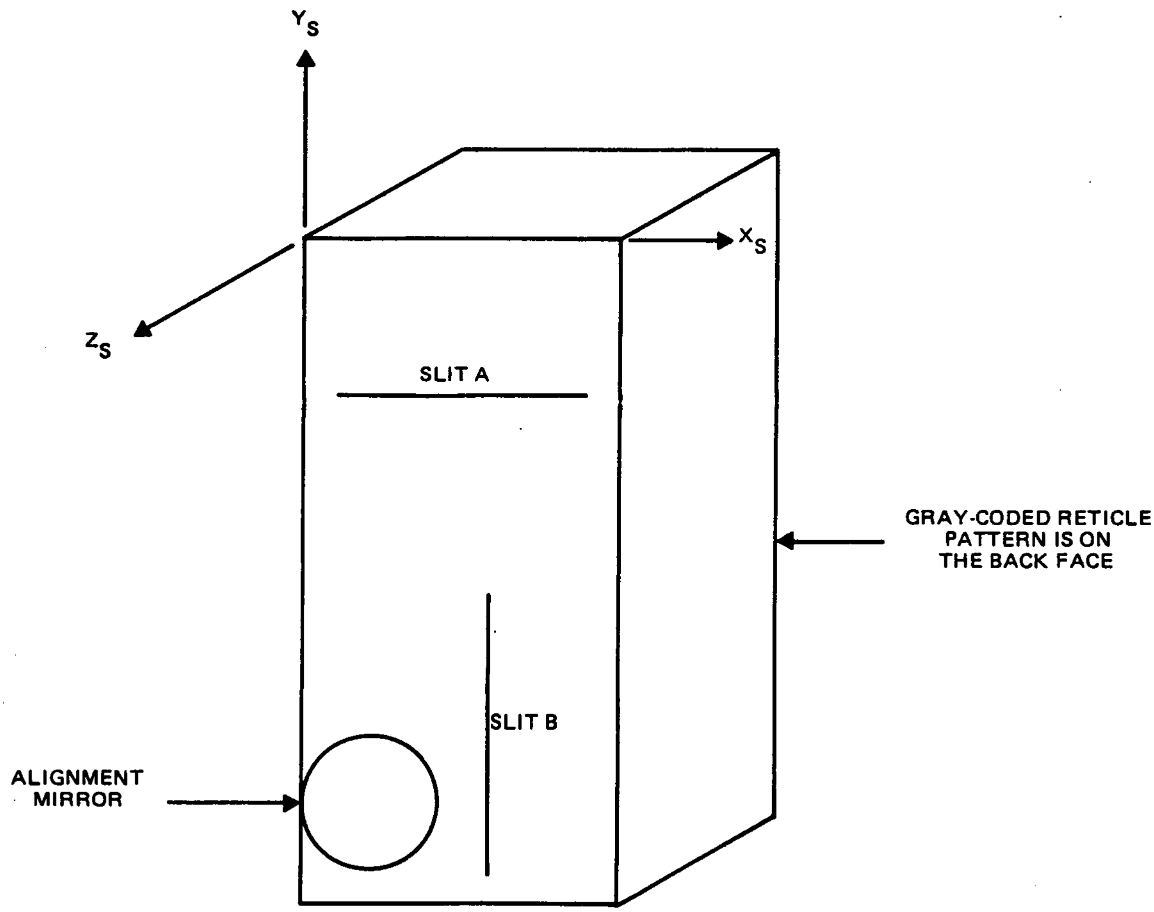


Figure 7-7. Definition of Two-Axis Sensor Reference Axes

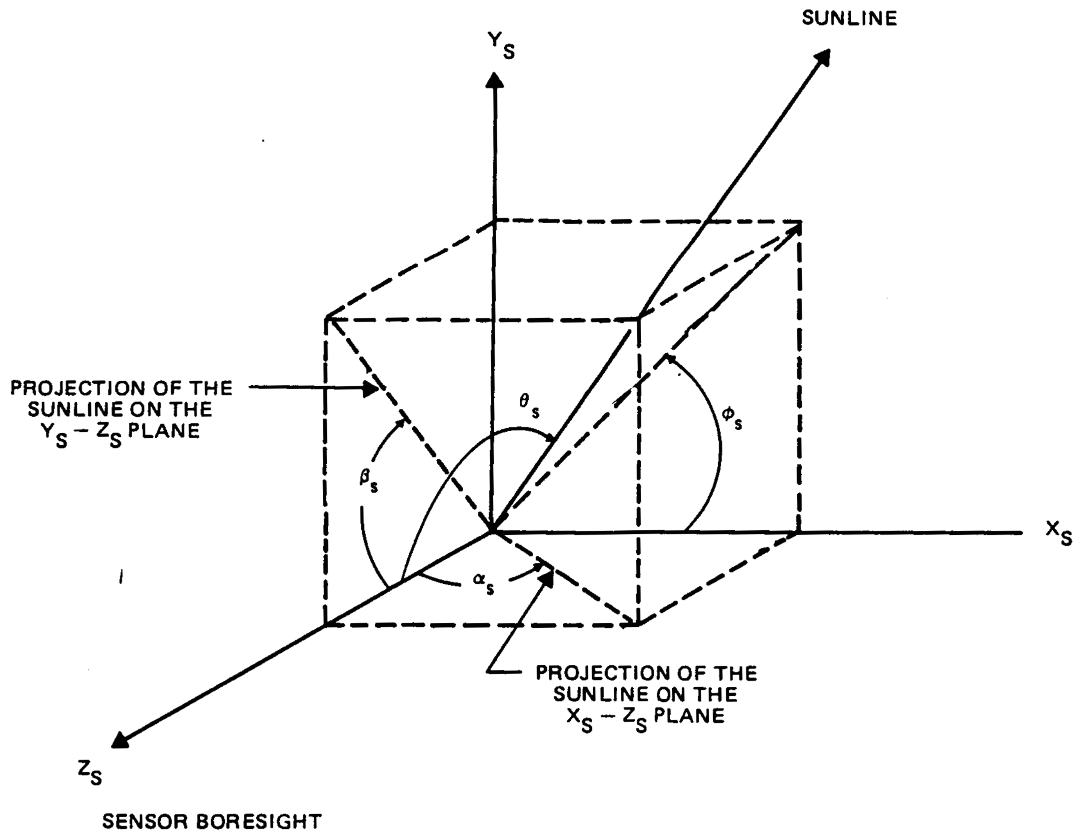


Figure 7-8. Two-Axis Digital Sun Sensor Reference Angles

The three most significant bits are as follows:

Three Most Significant Bits	Selected Sensor Head
001	1
010	2
011	3
000	None

All other codes signify invalid Sun sensor data.

Bits 4 through 11 and 12 through 19 are NA and NB, respectively, in Gray code. The conversion to Gray code is performed as follows:

1. Express NA (or NB) in binary code.
2. Bit 4 (or 12) is the most significant binary bit of NA (or NB).
3. Each succeeding Gray bit is the complement of the corresponding binary bit if the preceding binary bit is '1' or is the same if the preceding binary bit is '0'.

Example: NA = 61 (0 0 1 1 1 1 0 1 binary)

NB = 108 (0 1 1 0 1 1 0 0 binary)

ID = 2

The encoded bit stream is

Most Significant Bit $\qquad\qquad$ Least Significant Bit

$$\underbrace{0\,1\,0}_{\text{ID}}\quad \underbrace{0\,0\,1\,0\,0\,0\,1\,1}_{\text{NA}}\quad \underbrace{0\,1\,0\,1\,1\,0\,1\,0}_{\text{NB}}$$

7.4.3 Infrared Horizon Scanner

Let $\hat{S}_R$ denote the spin axis of the wheel, which is assumed fixed in inertial space for one revolution, and $\bar{E} = (E_x, E_y, E_z)^T$ the Earth to spacecraft vector, with the components given in kilometers. The lens prism is modeled as a rotating mirror mounted on the spin axis, whose normal, $\hat{n}$, rotates about the spin axis; the angle between $\hat{S}_R$ and the mirror normal is constant, $\gamma/2$ degrees, where γ is the scanner half-cone angle (γ = 135 degrees). The position of the mirror at any time is defined by a dihedral angle, φ, about the spin axis from the mirror normal, to a fixed direction (in this case, the spacecraft to Earth vector, $-\bar{E}$) (see Figure 7-9).

An orthogonal triad is defined to determine the position of the mirror normal in the geocentric inertial (G.I.) system. Define the vectors $\hat{S}_1$, $\hat{S}_2$, and $\hat{S}_3$ by

$$\hat{S}_1 = \hat{S}_R$$

$$\hat{S}_2 = \frac{\hat{S}_R \times \hat{E}}{|\hat{S}_R \times \hat{E}|} \tag{7-47}$$

$$\hat{S}_3 = \hat{S}_1 \times \hat{S}_2$$

Thus, in G.I. coordinates the mirror normal is

$$\hat{n} \equiv \hat{n}(\varphi) = \hat{S}_1 \cos\frac{\gamma}{2} + \sin\frac{\gamma}{2}\,(-\hat{S}_2 \sin\varphi + \hat{S}_3 \cos\varphi) \tag{7-48}$$

Let $\hat{b}_1$ and $\hat{b}_2$ be unit vectors denoting the position of the two bolometer flakes. It is assumed that $\hat{b}_1$, $\hat{S}_1$, and $\hat{b}_2$ are coplanar and that the angular separation between each flake and the spin axis is σ. In scanner coordinates the position of flake i (i = 1 or 2) is

$$\hat{b}_i = (-\sin\sigma_i \cos\eta_i,\ \sin\sigma_i \sin\eta_i,\ \cos\sigma_i)^T \tag{7-49}$$

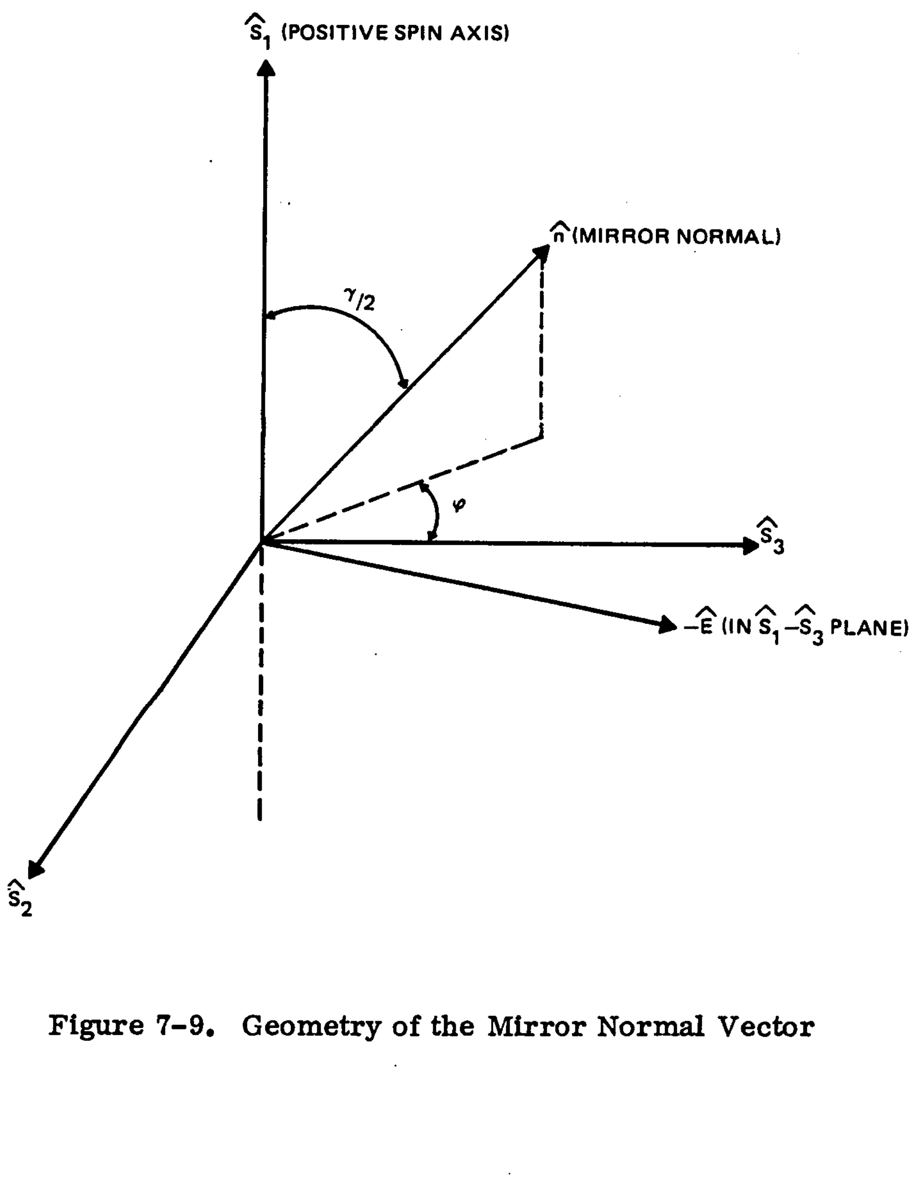


Figure 7-9. Geometry of the Mirror Normal Vector

The nominal values (for pitch zero) are

$$\sigma_1 = \sigma_2 = 3.5 \text{ degrees} \tag{7-50}$$

and

$$\begin{aligned} \eta_1 &= 0 \text{ degrees} \\ \eta_2 &= 180 \text{ degrees} \end{aligned} \tag{7-51}$$

The orientation of the flakes in the body is shown in Figure 7-10, where $\hat{X}_{IR}$ is assumed to be along the spacecraft's Z-axis.

Due to the symmetry of the incoming and outgoing rays from the mirrors, the bolometer will detect a horizon crossing when a ray emitted from the location of the bolometer flake is reflected on the mirror such that the reflected ray is tangent to the ellipsoid of the Earth.

For a bolometer located at $\hat{b}$ and a mirror with normal $\hat{n}$, the reflected (outgoing) ray from the spacecraft to the Earth (see Figure 7-11) is

$$\begin{aligned} \hat{x}_1 &= 2\hat{n}\,(\hat{b}_1 \cdot \hat{n}) - \hat{b}_1 \\ \hat{x}_2 &= 2\hat{n}\,(\hat{b}_2 \cdot \hat{n}) - \hat{b}_2 \end{aligned} \tag{7-52}$$

The equation of the ray from the Earth's geocenter to the point of tangency is

$$\bar{x} = \bar{E} + \alpha \hat{x}_1 \tag{7-53}$$

where $\alpha = (-\infty, \infty)$, and $\hat{x}_1 = (x_1, y_1, z_1)^T$ in G.I. coordinates.

The intersection of the ray, Equation (7-53), with the thermal horizon is computed as follows.

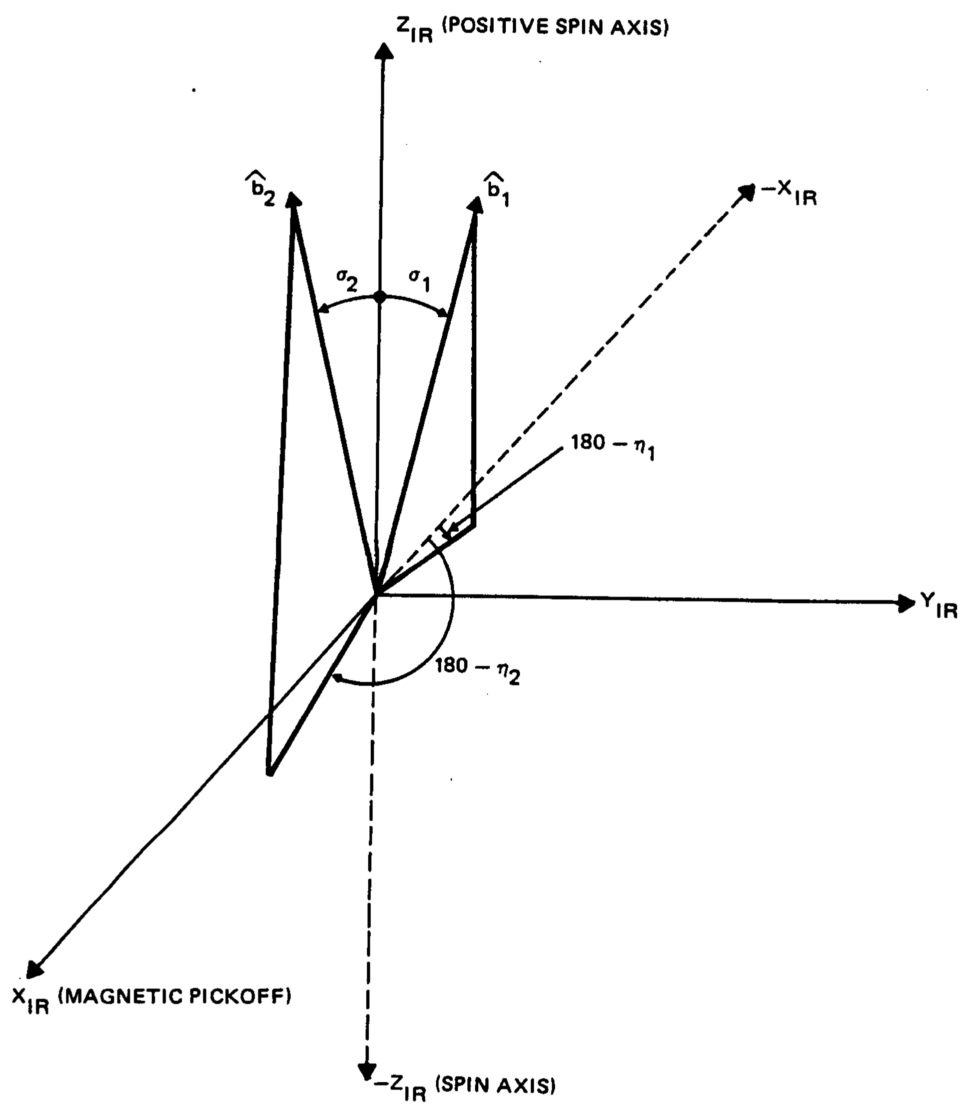


Figure 7-10. Geometry of Flakes in Scanner Coordinates

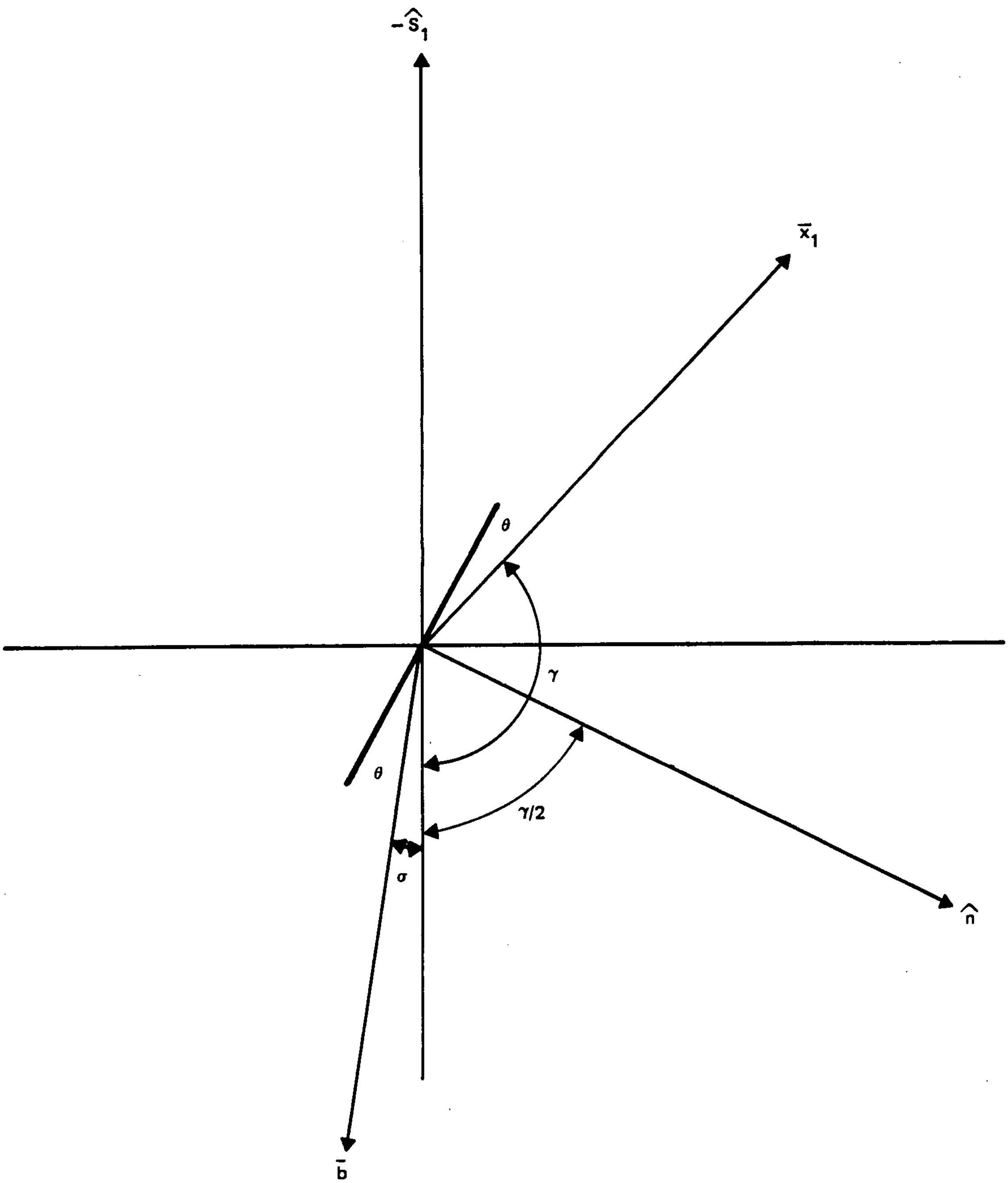


Figure 7-11. Geometry of Mirror and Reflected Ray

Let [A] be the attitude matrix from inertial to body coordinates and [W] the transformation matrix from the scanwheel reference frame to body coordinates. The scanwheel reference frame is defined by the wheel axis $(\hat{Z}_{IR})$ and pitch reference or magnetic pickoff $(\hat{X}_{IR})$. The [W] matrix is

$$[W] = \left[\hat{X}_{IR} \; \vdots \; \hat{Z}_{IR} \times \hat{X}_{IR} \; \vdots \; \hat{Z}_{IR} \right] \tag{7-54a}$$

or

$$[W] = \begin{bmatrix} 0 & -1 & 0 \\ 0 & 0 & -1 \\ 1 & 0 & 0 \end{bmatrix} \tag{7-54b}$$

where $\hat{X}_{IR} \cdot \hat{Z}_{IR} = 0$ and, nominally, the scanner angular momentum is along the spacecraft negative Y-axis and the magnetic pickoff is along the spacecraft Z-axis.

In inertial coordinates, the scanner axis of rotation is

$$\hat{S}_1 = [A]^T [W] \begin{bmatrix} 0 \\ 0 \\ 1 \end{bmatrix} \tag{7-55}$$

The horizon crossing vectors satisfy the cones equations

$$\begin{aligned} \hat{H} \cdot \hat{S}_1 &= \cos \gamma \\ -H \cdot \hat{E} &= \cos \rho \\ \hat{H} \cdot \hat{H} &= 1 \end{aligned} \tag{7-56}$$

Equation (7-56) assumes that the Earth is a sphere of angular radius ρ and that the scanner has a single flake with an effective scan angle of γ (nominally 135 degrees) relative to the scanner rotation axis. The error introduced into the scanner model by these approximations is negligible because only the approximate latitude of the horizon crossings is computed.

The latitudes corresponding to AOS(I) and LOS(O) are

$$\lambda_{I,O} = 90^{o} - \cos^{-1} (R_{I,O} (3)) \tag{7-57}$$

where $R_{I,O}(3)$ denotes the third component of the vector

$$\hat{R}_{I,O} = \hat{H}_{I,O} / \tan\rho + \hat{E} / \sin\rho \tag{7-58}$$

and $\hat{H}_{I,O}$ are the two solutions to the quadratic cones equations (Equation (7-56)).

Let the effective height of the CO_2 atmosphere as seen by the sensor at latitude $\lambda_{I,O}$, and the appropriate season be denoted by

$$h_{I,O} \equiv h\,(\lambda_{I,O},\ t) \approx 40 \text{ kilometers} \tag{7-59}$$

Then the local approximation to the Earth's shape as seen by the IR scanner at the latitude $\lambda_{I,O}$ is the ellipsoid

$$\frac{x^2 + y^2}{a^2_{I,O}} + \frac{z^2}{c^2_{I,O}} = 1 \tag{7-60}$$

where $a_{I,O} = 6378.14 + h_{I,O}$ and $c_{I,O} = 6356.755 + h_{I,O}$ are constant.

The intersection of the line $\bar{x}$ (Equation (7-53)) with the ellipsoid (Equation (7-60)), yields

$$\frac{(E_x + \alpha x_1)^2 + (E_y + \alpha y_1)^2}{a_{I,O}^2} + \frac{(E_z + \alpha z_1)^2}{c_{I,O}^2} = 1 \qquad (7\text{-}61)$$

which is a second-degree equation for α:

$$\left[\frac{x_1^2 + y_1^2}{a_{I,O}^2} + \frac{z_1^2}{c_{I,O}^2}\right]\alpha^2 + 2\left[\frac{E_x x_1 + E_y y_1}{a_{I,O}^2} + \frac{E_z z_1}{c_{I,O}^2}\right]\alpha + \left[\frac{E_x^2 + E_y^2}{a_{I,O}^2} + \frac{E_z^2}{c_{I,O}^2} - 1\right] = 0 \qquad (7\text{-}62)$$

or

$$A\alpha^2 + 2B\alpha + C = 0 \qquad (7\text{-}63)$$

where

$$A = \frac{x_1^2 + y_1^2}{a_{I,O}^2} + \frac{z_1^2}{c_{I,O}^2} \qquad (7\text{-}64)$$

$$B = \frac{E_x x_1 + E_y y_1}{a_{I,O}^2} + \frac{E_z z_1}{c_{I,O}^2} \qquad (7\text{-}65)$$

$$C = \frac{E_x^2 + E_y^2}{a_{I,O}^2} + \frac{E_z^2}{c_{I,O}^2} - 1 \qquad (7\text{-}66)$$

$\hat{x}$ can be determined for a given $\hat{S}_1$, pitch, φ, and flake from Equations (7-47) to (7-52), thus fixing A, B, and C.

The scanwheel axis attitude, $\hat{S}_1$, is given by Equation (7-55) and the true pitch angle in scanner coordinates is given by

$$p = \text{ATAN2}\,(a, b) \tag{7-67}$$

where

$$a = -\hat{Z}_{IR} \cdot (\hat{E}_B \times \hat{X}_{IR}) \tag{7-68a}$$

$$b = \hat{E}_B \cdot \hat{X}_{IR} \tag{7-68b}$$

The solution of Equation (7-63) is

$$\alpha = \frac{-B \pm \sqrt{B^2 - AC}}{A} = \frac{-B \pm \sqrt{D}}{A} \tag{7-69}$$

where

$$D = B^2 - AC \tag{7-70}$$

The line $\bar{x}$ is tangent to the Earth if $D = B^2 - AC = 0$. Those angles φ for which

$$D(\varphi) = 0 \tag{7-71}$$

are next determined. For a given set of $\hat{b}$, $\hat{S}_1$, and $\bar{E}$, there are, in general, two angles satisfying Equation (7-71), corresponding to acquisition of signal (AOS) and loss of signal (LOS). However, for $\lambda = \lambda_I$, the solution corresponding to AOS should be selected, and for $\lambda = \lambda_O$, the solution corresponding to LOS should be selected. For the solution of this nonlinear equation, an iterative scheme can be used by applying the method of regula falsi (Reference 12).

It is to be noted that if there is any solution to Equation (7-71), certainly the ray which lies in the plane defined by the spin axis and the spacecraft to Earth vector must intersect the Earth; i.e., for $\varphi_1 = 0$, $D_1 \equiv D(\varphi_1) > 0$. Also, because the subtended half-angle of the Earth must be less than 90 degrees, $D_2 \equiv D(\varphi_2) < 0$ for $\varphi_2 = \pm 90$ degrees. The solution φ_I, corresponding to AOS, satisfies -90 degrees $\leq \varphi_I \leq 0$ and the solution φ_O, corresponding to LOS, satisfies $0 \leq \varphi_O \leq 90$ degrees. Thus for $\lambda = \lambda_I$, set $\varphi_2 = -90$ degrees, and for $\lambda = \lambda_O$, set $\varphi_2 = 90$ degrees. The iterative procedure is

$$\varphi_{k+1} = \frac{\varphi_k D_\ell - \varphi_\ell D_k}{D_\ell - D_k} \tag{7-72}$$

where

$$D_{k+1} \equiv D(\varphi_{k+1}) \tag{7-73}$$

and ℓ is the largest subscript less than k for which the sign of D_ℓ is opposite that of D_k. This procedure always converges, because $D_\ell - D_k \neq 0$ and φ_{k+1} lies between φ_k and φ_ℓ (see Figure 7-12).

The iteration can be terminated whenever any of the following conditions are satisfied:

$$|\varphi_{k+1} - \varphi_k| \leq \epsilon_1 \tag{7-74a}$$

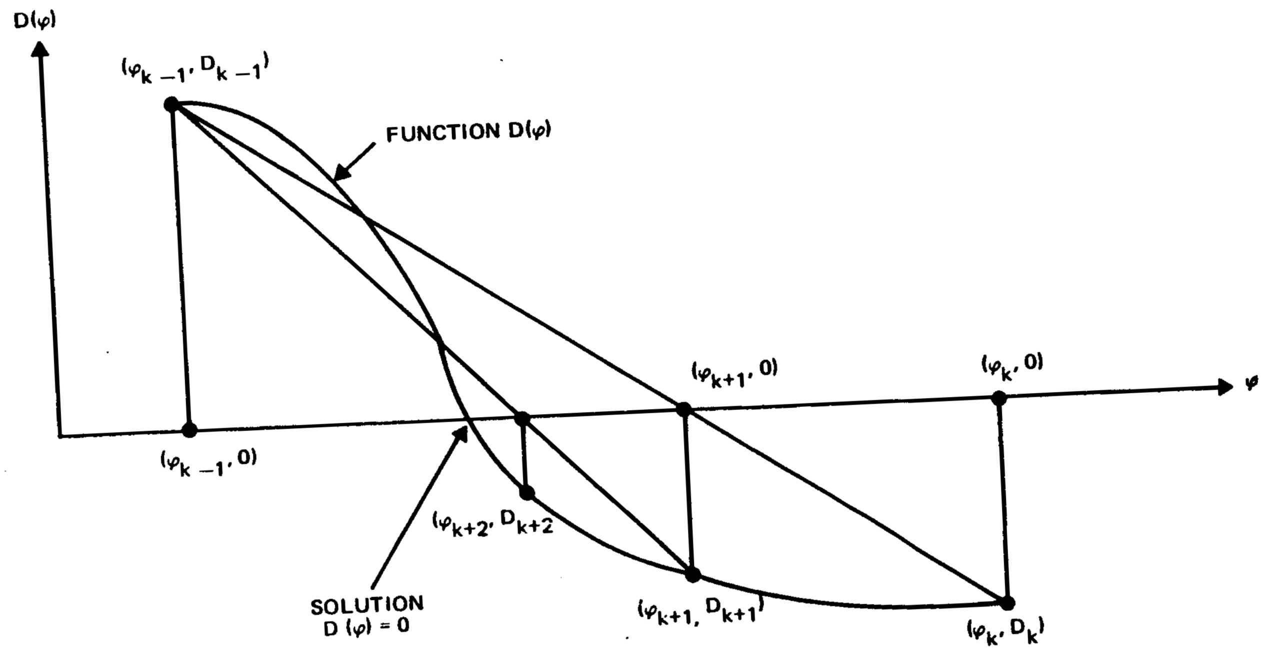


Figure 7-12. Method of Regula Falsi

$$|D_{k+1}| \leq \epsilon_2 \tag{7-74b}$$

$$k \geq N \tag{7-74c}$$

This procedure (Equations (7-71) through (7-74)) yields two solutions, φ_I and φ_O, satisfying Equation (7-71) for λ_I and λ_O, respectively. These dihedral angle solutions are computed for both flakes of the bolometer, yielding φ_{I1} and φ_{O1} from flake 1 and φ_{I2} and φ_{O2} from flake 2.

The dual flake logic selects one value for each dihedral angle by

$$\varphi_I = \max(\varphi_{I1}, \varphi_{I2}) < 0 \tag{7-75a}$$

$$\varphi_O = \min(\varphi_{O1}, \varphi_{O2}) > 0 \tag{7-75b}$$

where the algorithm, Equations (7-72) to (7-74), ensures that φ_{I1}, $\varphi_{I2} < 0$ and φ_{O1}, $\varphi_{O2} > 0$.

The scanner pitch, roll, and duty cycle output may be determined by considering the sequence of events during a single revolution of the scanwheel as shown in Figure 7-13.

The sequence is initiated 180 degrees away from the magnetic pickoff resulting in four possible sequences of events:

- <u>Case 1</u>: AOS and LOS data occur before the magnetic pickoff pulse.
- <u>Case 2</u>: AOS occurs before and LOS occurs after the magnetic pickoff pulse (scanner pitch near zero).
- <u>Case 3</u>: Both AOS and LOS occur after the magnetic pickoff pulse but before completion of the scanwheel revolution.

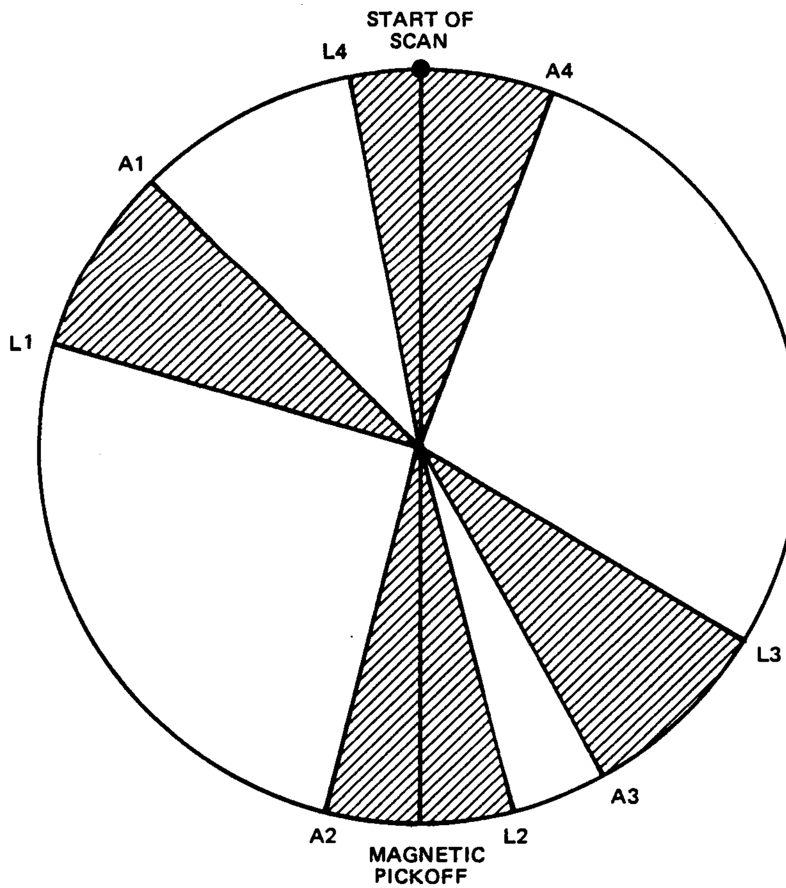


Figure 7-13. Possible Sequence of Events of a Scanwheel Revolution

- Case 4: LOS occurs before and AOS occurs after the magnetic pickoff pulse (scanner pitch near 180 degrees, i.e., the target is in the FOV at the beginning of the scanwheel revolution).

In all four cases, a duty cycle in-angle, H_I, and out-angle, H_O, are defined. For Case 1, H_I is the angle between AOS and LOS, and H_O is zero. For Case 2, H_I is the angle between AOS and the magnetic pickoff pulse and H_O is the angle between the magnetic pickoff pulse and LOS. For Case 3, H_I is zero and H_O is the angle between AOS and LOS. For Case 4, H_I is the angle between the start of the scan cycle and LOS, and H_O is the angle between AOS and the start of the next scan cycle.

To determine which of the four cases applies for a particular revolution of the scanwheel, it is necessary to examine the angular separation between $\hat{S}_3$ and the magnetic pickoff (see Figure 7-14).[1] The azimuth of $\hat{S}_3$ in scanner coordinate, α_3, is

$$\alpha_3 = \text{ATAN2}\,(S'_2, S'_1) \tag{7-76a}$$

where $\hat{S}' = (S'_1, S'_2, S'_3)^T \equiv [W]^T [A] \hat{S}_3$

For $-\pi \leq \alpha_3 < \pi$, the conditions are

- Case 1:

$$-\alpha_3 > \varphi_O$$

and

$$0 < -\alpha_3 < \pi + \varphi_I \tag{7-76b}$$

[1] p and p' are negative in Figure 7-14.

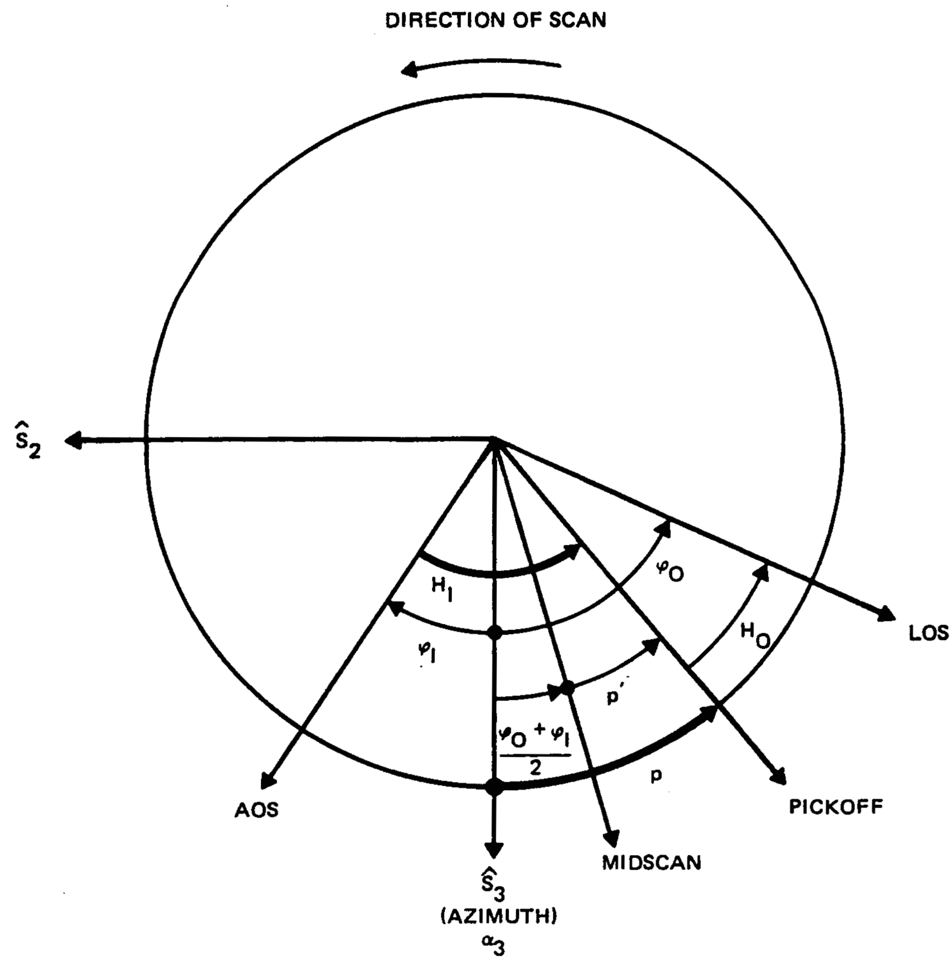


Figure 7-14. Geometry for the Computation of Scanner Data

- Case 2:

$$-\varphi_I \geq \alpha_3 \geq 0$$

or (7-76c)

$$-\varphi_O \leq \alpha_3 \leq 0$$

- Case 3:

$$\alpha_3 > - \varphi_I$$

and (7-76d)

$$0 < \alpha_3 \leq \pi - \varphi_O$$

- Case 4:

$$0 \leq - \alpha_3 \geq \pi + \varphi_I$$

or (7-76e)

$$0 \leq \alpha_3 \geq \pi - \varphi_O$$

The duty cycle in- and out-angles are related to the dihedral angles φ_I and φ_O as follows:

- Case 1:

$$H_I = \varphi_O - \varphi_I$$

$$H_O = 0 \qquad (7\text{-}77a)$$

- <u>Case 3</u>:

$$H_I = 0$$
$$H_O = \varphi_O - \varphi_I \qquad (7\text{-}77b)$$

Figure 7-14 describes the geometry for Case 2, when AOS precedes and LOS follows the magnetic pickoff pulse. The nadir vector is contained in the $\hat{S}_1 - \hat{S}_3$ plane, where the axis of rotation, $\hat{S}_1$, is out of the plane of the figure. The true pitch angle, p , is the angle from $\hat{S}_3$ to the spacecraft Z-axis (the figure depicts the situation where the spacecraft Z-axis and the magnetic pickoff are aligned, i.e., a pitch zero maneuver). Then

$$H_I = p - \varphi_I$$
$$H_O = \varphi_O - p \qquad (7\text{-}77c)$$

For Case 4, a similar analysis yields the result

$$H_I = \varphi_O - p + \text{sign}\ (\pi,\ p)$$
$$H_O = p - \varphi_I - \text{sign}\ (\pi,\ p) \qquad (7\text{-}77d)$$

The duty cycle in- and out-angles are converted to a voltage by

$$V_{AOS} = a_{VA}\ H_I + b_{VA}$$
$$V_{LOS} = a_{VL}\ H_O + b_{VL} \qquad (7\text{-}78a)$$

where for H_I and H_O in the range of 0 to 180 degrees (0 to 50 percent), the nominal parameters are

$$a_{VA} = a_{VL} = 0.0278 \text{ volts per degree}$$
$$b_{VA} = b_{VL} = 0 \qquad (7\text{-}78b)$$

Both V_{AOS} and V_{LOS} range from 0 to 5 volts.

The transmitted duty cycle data are

$$N_{AOS} = a_{AOS} V_{AOS} + b_{AOS}$$
$$N_{LOS} = a_{LOS} V_{LOS} + b_{LOS} \qquad (7\text{-}79)$$

where the nominal values for the parameters are

$$a_{AOS} = a_{LOS} = 51.2 \text{ counts per volt}$$
$$b_{AOS} = b_{LOS} = 0 \qquad (7\text{-}80)$$

From Figure 7-14, it can be seen that the scanner pitch angle, p' , is given by

$$p' = \frac{(H_O - H_I)}{2} \qquad (7\text{-}81)$$

The transmitted fine and coarse pitch errors have a range of ±2 and ±50 degrees, respectively, and are transmitted in two 8-bit words,

$$N_{PEF} = a_{PEF} (V_{LOS} - V_{AOS}) + b_{PEF} \tag{7-82a}$$

$$N_{PEC} = a_{PEC} (V_{LOS} - V_{AOS}) + b_{PEC} \tag{7-82b}$$

The voltage difference satisfies the inequality

$$-5 \leq V_{LOS} - V_{AOS} \leq 5 \tag{7-83}$$

which yields the following nominal values for the parameters:

$$\begin{aligned} a_{PEF} &= 1147.5 \text{ counts per volt} \\ a_{PEC} &= 45.9 \text{ counts per volt} \\ b_{PEF} = b_{PEC} &= 127.5 \text{ counts} \end{aligned} \tag{7-84}$$

The coarse and fine roll errors are transmitted in two 8-bit words

$$N_{REF} = a_{REF} (V_{AOS} + V_{LOS}) - b_{REF} \tag{7-85a}$$

$$N_{REC} = a_{REC} (V_{AOS} + V_{LOS}) - b_{REC} \tag{7-85b}$$

Assuming that $H_I + H_O = 107.6$ degrees ($V_{AOS} + V_{LOS} = 2.98$ volts) corresponds to zero roll; a sensitivity, $\Delta H/\Delta roll = 2.0$; and a full scale value of

±5 degrees for the fine error and ±50 degrees for the coarse error, the nominal parameters are

$$
\begin{aligned}
a_{REF} &= 459 \text{ counts per volt} \\
b_{REF} &= 1244.4 \text{ counts} \\
a_{REC} &= 45.9 \text{ counts per volt} \\
b_{REC} &= 9.69 \text{ counts}
\end{aligned}
\qquad (7\text{-}86)
$$

If N_{PEF}, N_{PEC}, N_{REF}, or N_{REC} are less than 0 or greater than 255, they assume the value of 0 or 255, respectively.

The pitch $(\Delta\theta)$, pitch rate $(\Delta\dot{\theta})$, and roll $(\Delta\varphi)$ error signals which are used by the control system are obtained from the duty cycle voltages by

$$\Delta\theta = V_{LOS} - V_{AOS} \qquad (7\text{-}87a)$$

$$\Delta\dot{\theta}_i = \frac{(\Delta\theta_{i-2} - 4\Delta\theta_{i-1} + 3\Delta\theta_i)}{2\Delta t} \qquad (7\text{-}87b)$$

$$\Delta\varphi = V_{AOS} + V_{LOS} - V_C \qquad (7\text{-}87c)$$

where Δt is the time step and V_C is the commanded roll voltage corresponding to the commanded roll bias, r_B (nominally $V_C = 2.98$ volts).

The procedure for modeling the scanwheel output is summarized in Table 7-3.

Table 7-3. IR Scanwheel Modeling Summary

STEP	EQUATION
1. COMPUTE ATTITUDE OF SCANWHEEL AXIS IN INERTIAL COORDINATES, $\hat{S}_1$, AND REFERENCE AXES, $\hat{S}_2$ AND $\hat{S}_3$	7–55 AND 7–47
2. COMPUTE NOMINAL TRIGGERING LATITUDES, $\lambda_{I,O}$	7–57
3. COMPUTE THE TRUE PITCH ANGLE, p	7–68
4. COMPUTE THE POSITION VECTOR OF THE BOLOMETER FLAKES, $\hat{b}_1$ AND $\hat{b}_2$	7–49 AND 7–51
5. COMPUTE THE REFLECTED RAYS FROM THE FLAKE, $\hat{x}_1$ AND $\hat{x}_2$	7–52
6. SET UP THE QUADRATIC EQUATIONS, FOR EACH REFLECTED RAY, FOR $\lambda = \lambda_I$	7–63 TO 7–66
7. DETERMINE TANGENT ANGLES, φ_{I1} AND φ_{I2} FOR EACH REFLECTED RAY USING THE ITERATIVE FALSI ALGORITHM	7–71 TO 7–74
8. REPEAT STEPS 6 AND 7 FOR $\lambda = \lambda_O$ DETERMINING φ_{O1} AND φ_{O2}	
9. COMPUTE THE DIHEDRAL ANGLES φ_O AND φ_I	7–75
10. COMPUTE THE AZIMUTH OF THE MAGNETIC PICKOFF, α_3, AND THE SCAN SEQUENCE OF EVENTS	7–76
11. COMPUTE THE DUTY CYCLE ANGLES, H_I AND H_O, AND VOLTAGES, V_{AOS} AND V_{LOS}	7–77 TO 7–78
12. COMPUTE THE DUTY CYCLE DATA, N_{AOS} AND N_{LOS}	7–79*
13. COMPUTE THE PITCH ERROR DATA, N_{PEF} AND N_{PEC}	7–82*
14. COMPUTE THE ROLL ERROR DATA, N_{REF} AND N_{REC}	7–85*
15. COMPUTE THE CONTROL SYSTEM ERROR SIGNALS, $\Delta\theta$, $\Delta\varphi$, AND $\Delta\dot{\theta}$	7–87

*SEE SECTION 4.2.1

7.4.4 Electromagnets

The three electromagnets are driven continuously by the onboard control system in the nominal range of ±10, 000 pole-centimeters. Information concerning the electromagnet status is transmitted in three 8-bit words,

$$\bar{N}_E = \begin{bmatrix} \text{Int}\,(C'_x D_{Cx} + 0.5) \\ \text{Int}\,(C'_y D_{Cy} + 0.5) \\ \text{Int}\,(C'_z D_{Cz} + 0.5) \end{bmatrix} \quad (7\text{-}88)$$

where $\bar{D}_C = (D_{Cx}, D_{Cy}, D_{Cz})^T$ is the commanded dipole field and C'_x, C'_y, C'_z are constants which are nominally 0.0128 counts per pole-centimeter.

7.4.5 Magnetic Field Rates

Magnetometer rate data is required by the control laws. Rate data may be simulated as follows. The magnetometer data, $\bar{M}'$, is obtained from Equation (7-38) and stored at each integration step. The rate data, $\dot{\bar{M}}'_i \equiv (\dot{M}'_x, \dot{M}'_y, \dot{M}'_z)^T$, in millioersteds per second at the ith step is

$$\dot{\bar{M}}_i = \left(\frac{\bar{M}'_{i-2} - 4\bar{M}'_{i-1} + 3\bar{M}'_i}{2\Delta t} \right) \quad (7\text{-}89)$$

where Δt is the time step.

The initial rate, $\dot{\bar{M}}'_0$, is assumed to be zero. The telemetered output is

$$\bar{N}_{BR} = \begin{bmatrix} \text{Int}\,(C''_x \dot{M}'_x + 0.5) \\ \text{Int}\,(C''_y \dot{M}'_y + 0.5) \\ \text{Int}\,(C''_z \dot{M}'_z + 0.5) \end{bmatrix} \quad (7\text{-}90)$$

where C''_x, C''_y, and C''_z are constants which are nominally $C''_x = C''_y = C''_z =$ 10.7 counts per millioersted per second to provide a full scale output of 12 millioersteds per second.

7.4.6 Wheel Speed

The 8-bit telemetered wheel speeds are assumed to be linearly related to the true wheel speed, S ,

$$N_S = a_S S + b_S \tag{7-91}$$

where for the coarse wheel speed, $0 \leq S < 2200$ rpm, the nominal values are

$$\begin{aligned} a_S &= 0.1164 \text{ counts per rpm} \\ b_S &= 0 \end{aligned} \tag{7-92}$$

7.4.7 Output Summary

Output from the attitude determination and control hardware is summarized in Table 7-4. The conversion from engineering units to volts to digital counts is the inverse of the algorithm described in Section 4.2.1.

In addition to the dynamically varying parameters described previously, the following data types are simulated and included in the MSOCC and IPD data records:

- Electromagnet bias
- A/D CONV CAL LEVEL 1
- A/D CONV CAL LEVEL 2
- A/D CONV CAL LEVEL 3
- 5-volt supply
- 10-volt supply

- MSOCC header data (Table 7-5)
- IPD header data (Table 7-7)
- Time and spacecraft status data (Table 7-6)

7.5 SYSTEMATIC AND RANDOM ERRORS

The subsection describes the various sources of errors which affect spacecraft data and outlines algorithms for simulation.

7.5.1 Electronic or Random Noise

Data types which are subject to random noise are listed in Table 7-4. The application of random noise to data requires one or more calls to a random number generator and consequently can be expensive in terms of CPU. Noise will be applied only on option. Uniformly distributed random numbers will suffice for most data types. Normally distributed random numbers may be generated

Table 7-4. Summary of Attitude Data

DATA TYPE	SYMBOL	WORDS	BITS PER WORD	DEFINING EQUATION
MAGNETOMETER†	$\bar{N}_V$	3	8	7-34††
SUN SENSOR IDENTIFICATION	ID	1	3*	7-43
SUN SENSOR ANGLE	NA	1	8*	7-44
SUN SENSOR ANGLE	NB	1	8*	7-44
PITCH ANGLE (FINE AND COARSE)†	N_{PEF} AND N_{PEC}	2	8	7-82††
DUTY CYCLES (AOS AND LOS)†	N_{LOS} AND N_{AOS}	2	8	7-79††
ROLL ANGLE ERROR (FINE AND COARSE)†	N_{REF} AND N_{REC}	2	8	7-85††
ELECTROMAGNET†	$\bar{N}_E$	3	8	7-88††
MAGNETIC FIELD RATE†	$\bar{N}_{BR}$	3	8	7-90††
WHEEL SPEED (FINE AND COARSE)†	N_S	1	8	7-91††

*FORMAT IS DESCRIBED IN APPENDIX A.

†SUBJECT TO RANDOM NOISE.

††THESE EQUATIONS ARE SCHEMATIC ONLY. THE DETAILED CONVERSION FROM ENGINEERING UNITS TO DIGITAL COUNTS IS THE INVERSE OF THE ALGORITHM DESCRIBED IN SECTION 4.2.1.

but require an order of magnitude more CPU time and consequently will be limited to the scanwheel data which requires precise modeling.

7.5.2 Random Bit Errors

The capability to invert randomly selected bits in the IPD 3492-byte or MSSOC 312-byte second will be provided. An input parameter will be provided for the bit error frequency--a nominal value is 10^{-6}. Assuming an MSOCC record length of $4 \times 312 = 1248$ bits, a bit error will occur on the average of once every 800 records. The inverted bit(s) is selected using a uniformly distributed random number generator.

7.5.3 Systematic Bit Errors

The capability to significantly increase the probability of random bit errors during specified intervals will be provided. This will effectively simulate transmission problems which might be expected to occur at the beginning or end of a real-time contact.

7.5.4 Time Tagging Errors

During specified intervals, the following two types of errors in the attached times will be simulated:

1. A specified constant attached time
2. Attached times displaced by a specified amount

Multiple intervals of time tagging errors will be accommodated.

7.5.5 Data Dropout

During specified intervals within a station pass, it should be possible to suppress the output of telemetry data to simulate data dropout.

7.6 DATA SETS

This subsection describes the input and output data sets required by the data simulator.

7.6.1 Input Data Sets

Input data sets are required for NAMELIST, ephemeris, and GESS nonresident tables.

7.6.1.1 NAMELIST

NAMELIST parameters will be provided in the following functional areas.

7.6.1.1.1 Initialization

The initial attitude and attitude rate may be expressed either in inertial coordinates, as a 3-1-2 Euler transformation (to specify the attitude of the wheel axis), or in orbital coordinates, as a 2-1-3 Euler transformation. In addition, provision for an initial attitude along the velocity vector is required. Input rate information may be either the body angular velocity or Euler angle rates.

7.6.1.1.2 Ephemeris

Spacecraft ephemerides will be input as either Keplerian orbital elements or an EPHEM data set as described in Section 7.6.1.2. The ephemeris input capabilities will be similar to that provided by subroutine EPHEMX (Reference 8).

7.6.1.1.3 Spacecraft and Hardware Parameters

The following input parameters are required to model the sensor output and spacecraft dynamics (symbols refer to parameters described in Sections 7.3 and 7.5):

- Sensor Output
  - Magnetometer alignment angles, θ_x, φ_x, θ_y, φ_y, θ_z, φ_z
  - Magnetometer bias field, $\bar{b}_B$
  - Magnetometer scale factors, $[P]$
  - Magnetometer null voltage, $\bar{V}_0$

- Magnetometer ADC factors, C_x , C_y , C_z
- Position vector of magnetometers relative to electromagnets, $\bar{r}_{EM}$
- Electromagnet compensation matrix $[C_D]$
- Sun sensor alignment angles, θ_i , ϕ_i , ψ_i , $i = 1, 2, 3$
- ATA scale factors, f_i , $i = 1, 2, 3$
- Sun visibility tolerance, ϵ
- Sun sensor threshold, cos 64
- Sun sensor scale and alignment parameters, a_0 , b_0 , k_a , k_b
- Sun sensor index of refraction, n
- Sun sensor thickness, h
- Effective mounting angle of IR scanner, γ
- Angular separation between each flake on spin axis, σ
- Alignment matrix and pitch maneuver flag for IR scanner, $[W]$ and k_p
- Effective CO_2 triggering altitude, $h_{I,O}$ (function of latitude and season)
- Polar and equatorial Earth radii, a , c
- Pitch angle coarse and fine calibration factors, a_{PEF} , a_{PEC} , b_{PEF} , b_{PEC}
- Duty cycle calibration factors, a_{LOS} , a_{AOS} , b_{AOS} , b_{LOS}
- Roll angle error coarse and fine calibration factors, a_{REF} , a_{REC} , b_{REF} , b_{REC}

- Electromagnet dipole ADC factors, C'_x , C'_y , C'_z
- Field rate ADC factors, C''_x , C''_y , C''_z
- Wheel speed ADC factors, a_S , b_S

- Spacecraft Dynamics
  - Spacecraft moments of inertia, I_1 , I_2 , I_3
  - Spacecraft residual dipole, $\bar{D}_R$
  - Commanded electromagnet null, $\bar{D}_{C0}$
  - Commanded roll bias, r_B
  - Solar radiation pressure torque, $\bar{\tau}_{SRP}$
  - Precession gain, K_1
  - Nutation gain, K_2
  - Acquisition gain, K_{SAT}
  - Unloading gain, K_{USAT}
  - Blanking factor, f_M
  - Pitch gains, K_θ , $K_{\dot{\theta}}$
  - Nominal wheel speed, S_w
  - Wheel moment of inertia, I_w
  - Wheel unloading deadbands, ΔS_{MAX} , ΔS_{TOL}
  - Wheel acceleration torque, τ_B
  - Wheel bearing torque
  - Control mode (acquisition/on-orbit)
  - Orbit adjust torque, $(\Delta\bar{r} \times \bar{F})$
  - Orbit adjust thrust application time, t_{OA}
  - Orbit adjust thrust duration, Δt_{OA}

7.6.1.1.4 Integrator and Simulation Control

The following parameters control the interval to be simulated and integrator. Input times should be in calendar format (YYMMDD.HHMMSS). Integrator parameters assume the use of the Adams-Bashforth integrator described in

Section 7.4; however, analogous parameters are required for a Runge-Kutta or Adams-Moulton integrator.

- Integrator start time, TSTART
- Integrator stop time, TSTOP
- Number of first-order equations, N
- Specified accuracy for each equation, $\bar{\epsilon}$
- Accuracy window for interval halving and doubling, N_D
- Output rate, T_{OUT}
- Initial step size, h_0
- Integration interval control, IFRC
- Telemetry output flags (disk/tape, MSOCC/IPD)
- Analytical output flag (hardcopy and/or data set)
- Debug output flags
- Station coverage time intervals
- Iteration parameters for regular falsi algorithm, ϵ_1, ϵ_2, and N
- Initial attitude state; format a, b, or c

  = a, pitch, roll, yaw, and rates
  = b, $\hat{Y}$-axis attitude and body rates
  = c, $\hat{Z}$-axis along velocity vector and body rates

7.6.1.1.5 Systematic and Random Error Control

The following parameters should be provided to simulate noise and random or systematic errors in the output analytical or telemetry output:

- Noise on each analog data type in Table 7-4
- Random bit error rate

- Time intervals for accelerated random bit rate errors
- Time tagging error type (constant or displaced)
- Time tagging error intervals
- Data dropout intervals

7.6.1.2 Ephemeris

Required ephemeris data sets include an EPHEM tape or disk and a solar/lunar SUNRD file or the equivalent.

7.6.2 Output Data Sets

Output data sets are required to simulate the POCC and IPD attitude telemetry data sets and provide simulated data to exercise the residual dipole bias determination and scanner bias determination utilities.

7.6.2.1 Operations Control Center (OCC) Telemetry Data Set

A data set, similar in both form and content to that provided by the OCC over the Attitude Data Link (ADL), is required. In addition to the attitude determination telemetry summarized in Table 7-4, header, attached time, spacecraft clock, and status bits shall be simulated. The MSOCC header information is summarized in Table 7-5 and the remaining data in Table 7-6. The data set format is described in Section 5.6.2.

7.6.2.2 IPD Telemetry Data Set

A data set similar in both form and content to that provided by IPD for definitive attitude support is required. Data in both the nominal orbital mode format and in the memory dump format, described in Appendix B, is simulated. The data set contains attitude determination telemetry (Table 7-4), time and status bit data (Table 7-6), and IPD header data (Table 7-7); its format is described in Section 5.6.2.

Table 7-5. MSOCC Header Data

DATA TYPE	BYTES PER TYPE	CONTENTS
SPACECRAFT ID	1	0000 1000 (08 HEX)
SPACECRAFT DATA MODE	1	0000 0000
DATA TYPE	1	0000 0000
TRANSMISSION RECORD NUMBER	2	BINARY INTEGER
STATION NAME	3	FOUR XDS 930 CHARACTERS
PASS CONTACT NUMBER	2	BINARY INTEGER
AOS	4	YYDDDHHMM (BINARY INTEGER)

Table 7-6. Time and Spacecraft Status Data

DATA TYPE	BYTES PER TYPE	BITS PER BYTE	CONTENTS (RANGE)
DAY OF YEAR	2	8	BINARY INTEGER
MILLISECONDS OF DAY	4	8	BINARY INTEGER
QUALITY FLAG	1	8	0000 0000 = BAD 0000 0001 = GOOD
MINOR FRAME COUNTER	1	8	BINARY INTEGER (1 – 8)
SPACECRAFT CLOCK	3	8	BINARY INTEGER
ROLL BIAS	1	8	BINARY INTEGER
PCM MODE WORD	1	8	BINARY INTEGER
STATUS FLAGS:			
ACQUISITION/ON-ORBIT	1	1	1 = ACQUISITION MODE
PITCH 0/±90 DEGREES	1	1	1 = ±90 DEGREES
MAGNETOMETER ON/OFF	1	1	1 = ON
MAGNETOMETER IN/OUT OF LOOP	1	1	1 = OUT
AUTOMATIC REACQUISITION IN/OUT	1	1	1 = IN
SUN SENSOR ON/OFF	1	1	1 = ON

Table 7-7. IPD Header Data

DATA TYPE	BYTES PER TYPE	CONTENTS
SPACECRAFT ID	1	1010 1000 (A8 HEX)
SPACECRAFT DATA MODE	1	0000 0000
DATA TYPE	1	0000 0001
RUNNING NUMBER OF RECORDS PER PASS	2	BINARY INTEGER
STATION NAME	3	FOUR XDS 930 CHARACTERS
RUNNING NUMBER OF RECORDS IN TRANSMISSION	2	BINARY INTEGER
DATE REPROCESSED AND NUMBER OF TIMES REPROCESSED	4	BINARY INTEGER
START TIME OF PASS	4	YYDDDHHMM (BINARY INTEGER)
STOP TIME OF PASS	4	YYDDDHHMM (BINARY INTEGER)

7.6.2.3 Utility and Analysis Data Sets

Data sets are required to test the pitch/roll scanner bias determination and residual magnetic dipole determination utilities. In addition, data sets are required for analysis and software testing. A general-purpose analytical data set containing simulated data in a digitzed engineering format, together with a truth model, will serve the following purposes:

- Testing the data adjustment and attitude determination subsystems prior to the completion of the telemetry processor subsystem
- Evaluating the performance and accuracy of various attitude determination and bias determination algorithms

Truth model information, the simulated attitude, and attitude rate should be included in the analytical data set. The formats of the four data sets, scanner bias utility source, wheel speed, electromagnet, and attitude history file are described in Sections 5.6.5, 5.6.4, 5.6.3, and 5.6.8, respectively. The format of the analytical data set is to be determined. The analytical output should include the following:

- Pitch, roll, yaw, and rates $(p, r, y, \dot{p}, \dot{r}, \dot{y})$
- Scanwheel axis attitude (α_y, δ_y)
- Angle between scanwheel axis and orbit normal (θ_{ny})
- Total angular momentum, magnitude, and direction in inertial coordinates (L, α_L, δ_L)
- Nutation half-cone angle $\left(\cos^{-1}(\hat{L} \cdot \hat{Y}) = \theta_N\right)$
- Sensor data in engineering units

SECTION 8 - UTILITY SPECIFICATIONS

8.1 LOG DATA SET INTERROGATION AND UPDATE UTILITY

The log data set interrogation and update utility is a standalone foreground program executed by the user via TSO. There are three basic functions which the utility provides:

- Display of data for evaluation by the analyst and update of the log data set
- Operator interrogation of the log data set and subsequent update of the transmission data set
- Operator update of the log data set after transmission of data to the IPD

The utility is used by analysts to perform quality assurance functions and communicate instructions to operators regarding the requested disposition of the data. The utility is used by the operator to archive information pertaining to data disposition.

The program reads a requested summary from the log data set and displays selected variables from the summary, or on option reads the entire data set, searching for and displaying selected variables from all summaries containing a requested flag value. This option allows the user to determine all data which have been processed but not reviewed by an analyst, all data reviewed by an analyst and found of insufficient quality for transmission to the IPD, all data of sufficient quality for transmission, or all data which have been transmitted to the IPD.

8.1.1 Utility Input

The input for the interrogation and update utility consists of the log data set, NAMELIST control parameters, and the attitude history file. The contents of the log data set are described in detail in Section 5.6 and include

- Header information (date and time span of the data, orbit number, etc.)
- Parameters indicating the quality of the attitude and attitude rate solutions
- Comments from operators or analysts regarding the solutions obtained
- Member name of the corresponding attitude solutions in the attitude history file
- Dates the data were processed, reviewed by an analyst, moved to the transmission data set, and received by the IPD, as well as the initials of the personnel who performed these functions
- A status flag indicating which functions have been performed on the data
- The tape and file numbers of the magnetic tape backing up the transmission data set

The NAMELIST parameters indicate

- Which summary or summaries are to be read
- Which variables, within the summary or summaries, are to be displayed

8.1.2 Utility Functional Description

The following functions are described individually because several capabilities of the utility are unique to each function.

8.1.2.1 Analyst Interrogation of the Log Data Set

Three options for displaying selected data from the log data set are provided to the analyst:

- Request a tabular display of selected variables from all summaries whose status flag is a particular value
- Request a tabular display of selected variables from all summaries for a given segment of data
- Request selected variables from a selected summary

The first option provides the analyst with a summary of all data processed since the last transmission to IPD and awaiting analyst review. The second option allows the analyst to determine how many summaries refer to a particular segment of data, i.e., whether any passes of data were reprocessed by the operator and are referred to by more than one summary.

The analyst invokes the third option on an individual summary-by-summary basis, determining in each case whether the attitude solutions described by the summary are of sufficient quality for subsequent transmission to the IPD. The value of the status flag in each summary is then altered to indicate that either the data are to be transmitted to the IPD, or the data have been reviewed and are not to be transmitted. The summary is updated to reflect these changes, and an appropriate field is automatically filled with the current date.

8.1.2.2 Operator Interrogation of the Log Data Set

The utility allows an operator to interrogate the log data set by displaying a tabular summary of the data which the analyst has reviewed and designated as ready for transmission. If there are any, the utility concatenates all such data into a (time-ordered) sequential transmission data set on a permanently mounted, sharable disk pack. This function is performed automatically by the utility if requested. The sequence of utility execution is

1. Search the log data set for all summaries with status flag values which indicate that the data is ready for transmission.

2. Check the start and stop times of the selected summaries and time-order the members.

3. Move the corresponding attitude solutions from the attitude history file on a member-by-member basis, based on the time-ordered sequence, to the sequential transmission data set. In so doing, the header information in the first record of each member becomes the 24-byte header of the IPD header record, and the attitude solution quality information contained in the following 64 bytes of the first record is used as the remainder of the IPD header record. The remaining attitude solution data in the first and subsequent records in the member are transferred accordingly. The number of the IPD record within the block (i.e., within the pass) is computed and inserted, as is the number of the IPD record within the transmission (i.e., within the attitude transmission data set being generated).

4. Update the status flag in each summary as the corresponding member in the attitude history file is moved to denote that the data have been placed in the transmission data set. Include in the summary the initials of the operator and insert the current date.

The operator then backs up the transmission data set on magnetic tape via a separate background job.

8.1.2.3 Operator Update of the Log Data Set

After the data in the transmission data set have been successfully transmitted to the IPD, the operator updates the status flag in the summaries corresponding to the data transmitted (this is done automatically on option), includes the transmission data set backup tape and file numbers, and enters his initials. The current date is automatically inserted.

8.1.3 Utility Output

The output of the interrogation utility consists of updated summaries written to the log data set, and hardcopy printout of these summaries. Output also includes data moved from the attitude history file to the transmission data set.

8.2 SCANNER BIAS UTILITY

The IR horizon scanner is subject to several sources of errors which affect attitude determination including mounting angle errors, electronic anomalies, calibration errors and bolometer offsets. If the geometry between the scanner and the Earth were to vary significantly, it would be necessary to solve for each source of error separately since each affects the data in different ways as the geometry changes. However, in on-orbit mode, the pitch and roll control laws drive the spacecraft to a nearly constant value of scanner duty cycle (commandable from the ground) and pitch angle. Consequently, the scanner biases are highly correlated and it would be very difficult to separate the effects of each source of error from the data. Therefore, it is advantageous to combine these errors into two observable parameters: a net pitch bias, p_B , and a net roll bias, r_B . The following analysis outlines an algorithm for determining these biases from IR scanner data, Sun sensor data, and ephemeris information.

8.2.1 Analytical Considerations

The orbital to spacecraft transformation [K] , expressed in terms of rotation angles p' , r' , and y' about the orbital pitch, roll, and yaw axis respectively, is defined in Equation (4.1-6). The true pitch and roll angles, p' and r' , are related to the scanner pitch and roll data, p and r , and the biases by

$$p' = p - p_B \qquad (8\text{-}1a)$$

$$r' = r - r_B \tag{8-1b}$$

The differential corrector algorithm DC, which is described in Section 4.4.4, is used to determine the pitch and roll biases. Using the notation of Section 4.4.4, the state vector is

$$\bar{X} = (p_B, r_B)^T \tag{8-2}$$

The measurement vector is

$$\bar{M} = (p, r, S_{B1}, S_{B2}, S_{B3})^T \tag{8-3}$$

where $\hat{S}_B \equiv (S_{B1}, S_{B2}, S_{B3})^T$ is the Sun unit vector in body coordinates. The orbital and body Sun unit vectors are related by

$$[K]^T \hat{S}_B = \hat{S}_O \tag{8-4}$$

The third component of this equation is

$$-\sin p' \cos r' S_{B1} + \sin r' S_{B2} + \cos r' \cos p' S_{B3} = S_{O3} \tag{8-5}$$

which is independent of the yaw angle.

Thus, the observation vector is chosen to be

$$\bar{Y} = S_{O3} \tag{8-6}$$

and the observation model vector is

$$f(\bar{X}, \bar{M}) = -\sin(p - p_B)\cos(r - r_B) S_{B1} + \sin(r - r_B) S_{B2} + \cos(r - r_B)\cos(p - p_B) S_{B3} \tag{8-7}$$

The matrix of partial derivatives which is required to evaluate Equation (4-106) is

$$\frac{\partial f(\bar{X}, \bar{M})}{\partial X_1} = \cos(p - p_B)\cos(r - r_B) S_{B1} + \cos(r - r_B)\sin(p - p_B) S_{B3}$$

$$\frac{\partial f(\bar{X}, \bar{M})}{\partial X_2} = -\sin(p - p_B)\sin(r - r_B) S_{B1} - \cos(r - r_B) S_{B2} + \sin(r - r_B)\cos(p - p_B) S_{B3} \tag{8-8}$$

8.2.2 Utility Input

Input to the scanner bias utility is obtained from NAMELIST and the scanner bias utility source data set. The contents of this data set, which are described in detail in Section 5.6, include

- IR scanner fine, coarse, and duty cycle generated pitch angles
- IR scanner fine, coarse, and duty cycle generated roll angles
- Sun unit vector in spacecraft coordinates, $\hat{S}_B$
- Sun unit vector in orbital coordinates, $\hat{S}_O$

NAMELIST parameters include

- Initial estimate of the fine, coarse, and duty cycle generated pitch biases
- Initial estimate of the fine, coarse, and duty cycle generated roll biases

- Start time of data to be processed
- Stop time of data to be processed
- Observation weights
- Maximum number of iterations
- FORTRAN unit number of scanner bias utility source data set
- FORTRAN unit number of scanner biases data set
- Saturation values for fine, coarse, and duty cycle generated pitch angles
- Saturation values for fine, coarse, and duty cycle generated roll angles
- FORTRAN unit number for hardcopy printout
- Maximum number of sensor data points
- Sun geometry criteria
- Observation mean tolerance level
- Observation standard deviation tolerance level
- Flag specifying whether or not to write results on scanner biases data set if the mean and standard deviation of the observation residuals are less than the tolerance level

Input displays for the scanner bias utility available on option consist of plots of the IR scanner fine, coarse, and duty cycle-generated pitch angles versus time and plots of the IR scanner fine, coarse, and duty cycle-generated roll angles versus time. Additional input plots are described in Figures 8-1 and 8-2.

8.2.3 Utility Functional Description

The scanner bias determination utility is a standalone program which may be operated in either interactive mode or batch mode under GESS. The utility

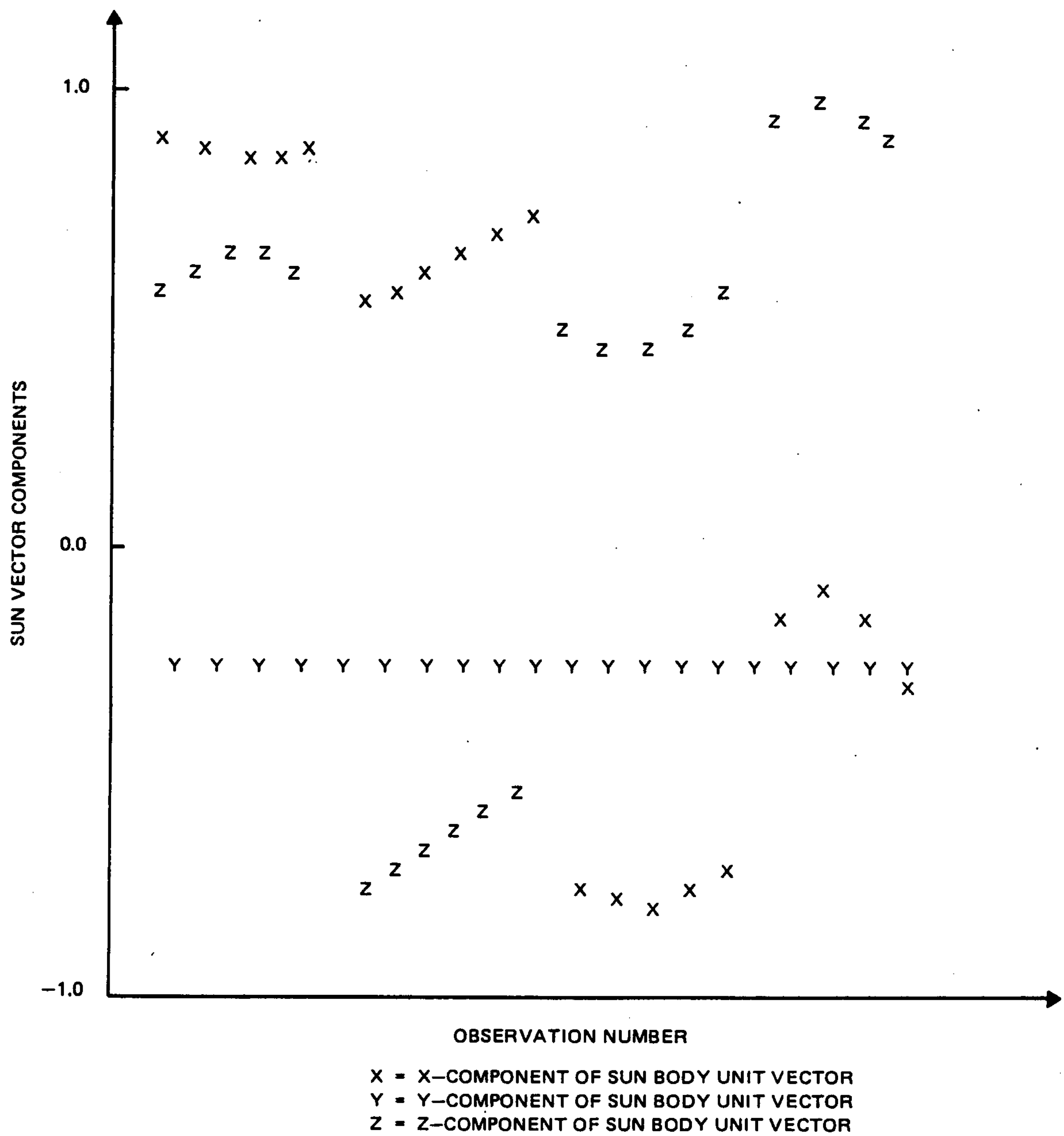


Figure 8-1. Plot of Sun Body Unit Vector Components

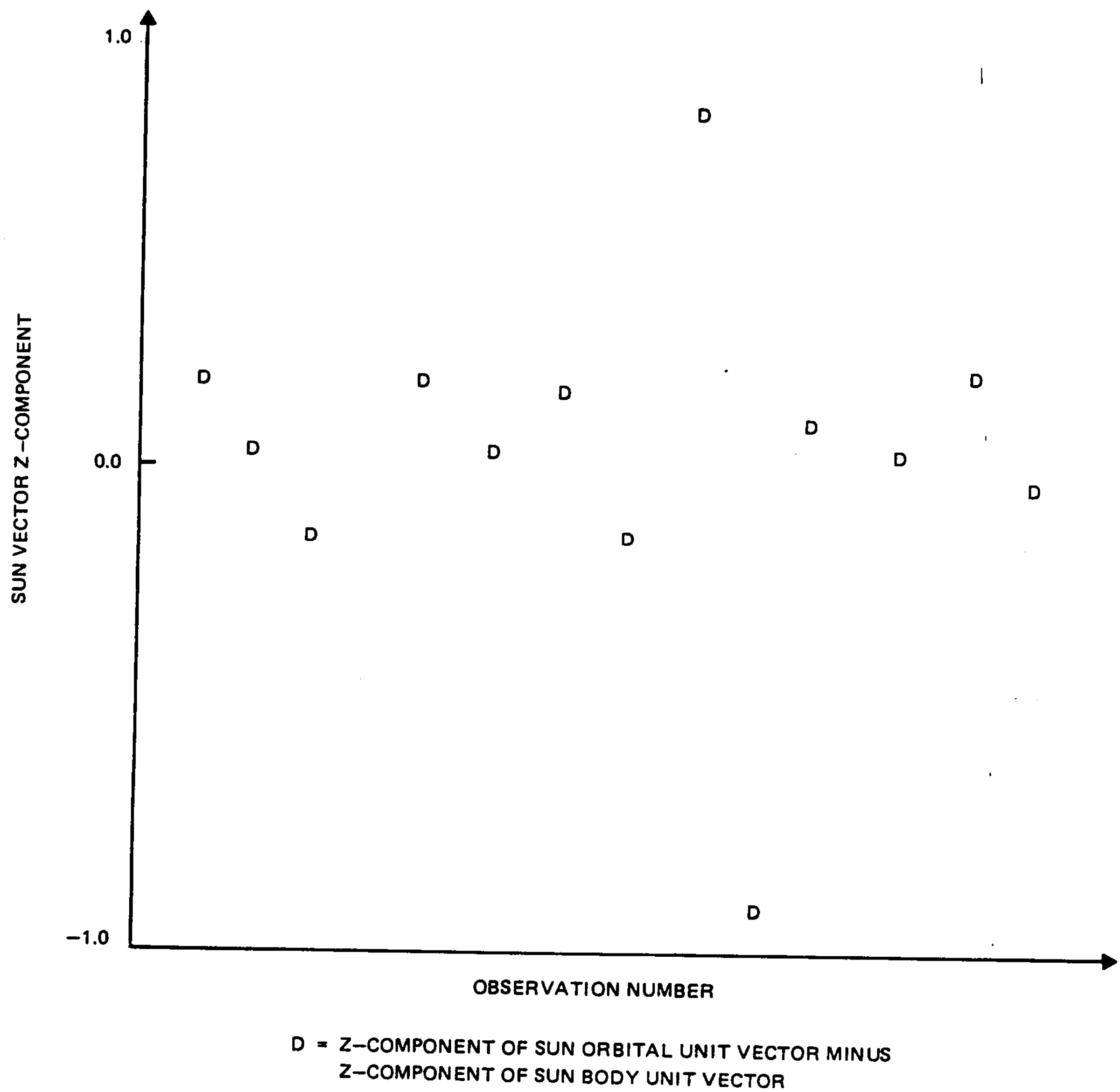


Figure 8-2. Plot of Difference of Sun Unit Vector Z-Components

first reads sensor data from the scanner bias utility source data set. All scanner data are rejected if they lie outside the processing interval established via NAMELIST or if the third component of the Sun unit vector in orbital coordinates is greater than some preassigned value (this means the Sun/spacecraft geometry is unfavorable for bias estimation). In addition, individual scanner data are rejected if they are equal to the saturation value. All remaining data are stored in arrays, up to some preassigned maximum number of data points.

The first iteration of the differential corrector produces an estimate of the pitch and roll biases and the mean and standard deviations of the observation residuals based on this estimate. If the values for the mean and standard deviation are less than the specified tolerances, no additional iterations are required. However, if the mean or standard deviation is greater than the tolerance value, another iteration is performed, a correction to the previous estimates is obtained, and new values for the mean and standard deviation are obtained. This cycle is repeated until both the mean and standard deviation are less than the tolerance level, or until the maximum number of iterations has been reached. After this processing is complete, the final estimates of the pitch and roll biases are written to the scanner biases data set provided that the observation residual mean and standard deviation are within the tolerance and that this output function is requested via NAMELIST. If the scanner biases are updated, the scanner bias increment number is increased by 1. Hardcopy printout of the estimated biases and their uncertanties, observation means, and observation standard deviations are generated at each iteration. Optionally, the observation residuals for each data point at each iteration may be printed.

8.2.4 Utility Output

If the pitch and roll biases have been successfully estimated, the output to the scanner biases data set consists of

- Start time of processing interval
- Stop time of processing interval

- Date processing was performed
- Estimate of fine, coarse, and duty cycle generated pitch biases
- Standard deviation of fine, coarse, and duty cycle generated pitch biases
- Estimate of fine, coarse, and duty cycle generated roll bias
- Standard deviation of fine, coarse, and duty cycle generated roll bias
- Scanner bias increment number

The data set disposition shall be SHR in the job control language and changed to MOD at execution time to permit concatenation of bias solutions.

Output displays for the scanner bias utility available on option consist of

- Plots of the residual versus observation number for each iteration
- Plots of the mean of the residuals versus iteration number
- Plots of the standard deviation of the residuals versus iteration number
- Plots of the estimated fine, coarse, and duty cycle-generated pitch biases versus iteration number
- Plots of the estimated fine, coarse, and duty cycle-generated roll biases versus iteration number

Additional output displays and plots are described in Figures 8-3, 8-4, and 8-5.

8.3 MAGNETIC DIPOLE BIAS DETERMINATION UTILITY

An apparent residual magnetic dipole on the HCMM spacecraft may be due not only to uncompensated spacecraft magnetic fields, but also to related sources, such as errors in the calibration curve relating electromagnet output to field strength and misalignments of the electromagnets. Because the HCMM orbit is nearly polar, the pitch component (Y-component in the body frame) of the

**** S C A N N E R B I A S T Y P E ****

ITER NO.	FINE PITCH	COARSE PITCH	D.C. PITCH	FINE ROLL	COARSE ROLL	D.C. ROLL
1	-0.404	-0.349	-0.503	1.252	1.451	-1.482
2	-0.322	-0.282	-0.418	1.098	1.248	-1.556
3	-0.344	-0.277	-0.434	1.107	1.266	-1.577
4	-0.339	-0.279	-0.429	1.111	1.273	-1.589

Figure 8-3. Scanner Bias Value Display

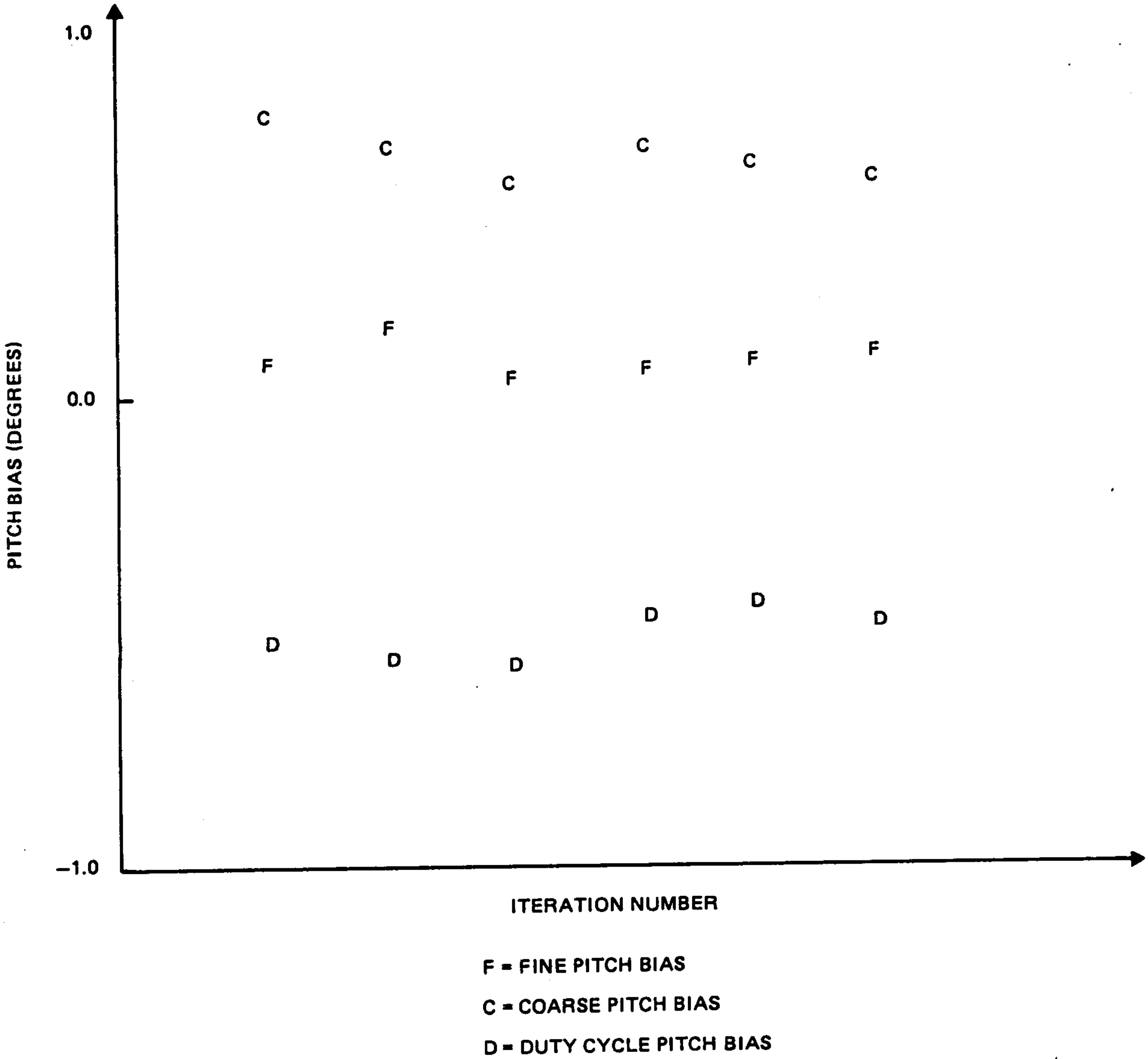


Figure 8-4. Plot of Estimated Pitch Biases

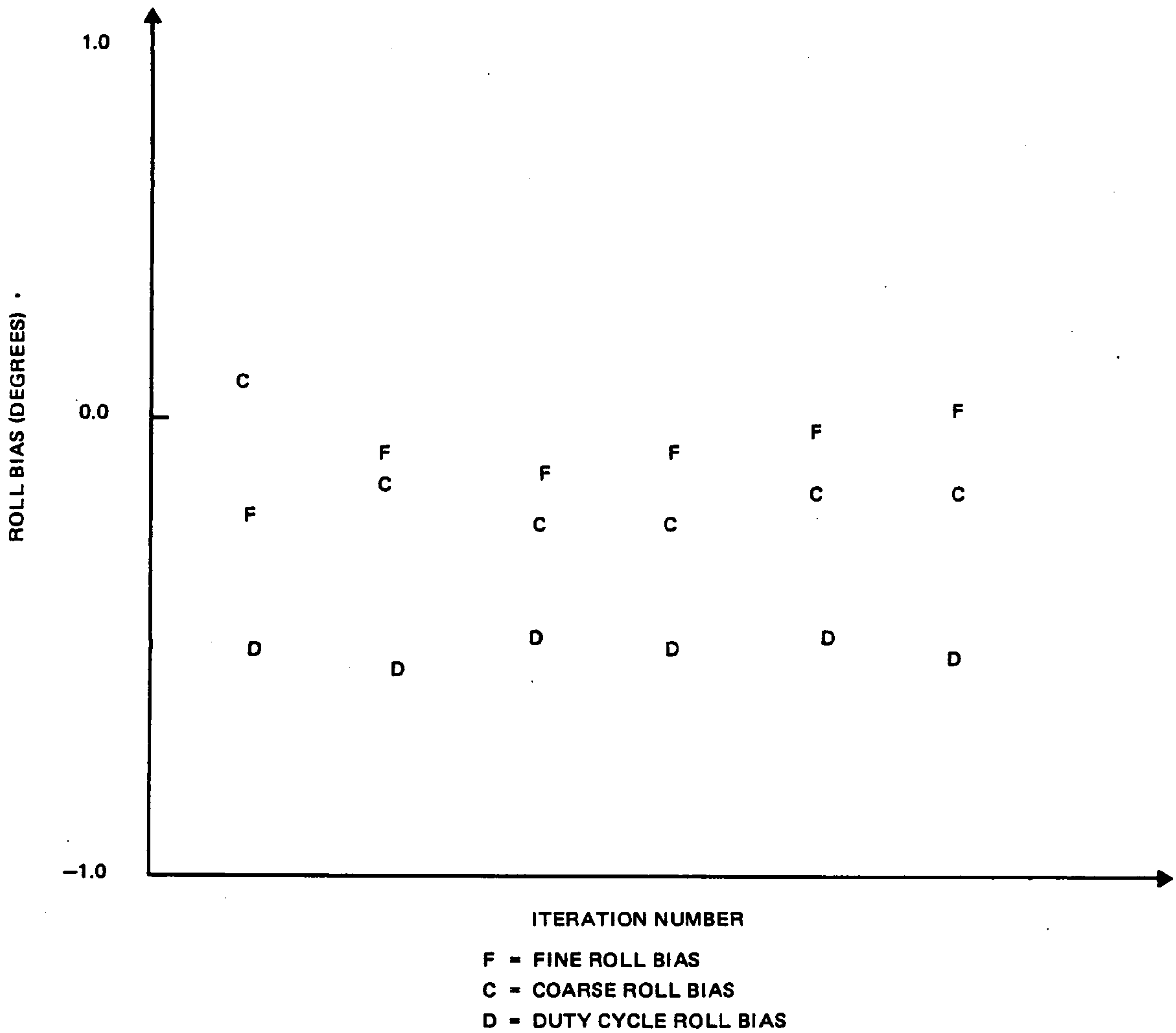


Figure 8-5. Plot of Estimated Roll Biases

Earth magnetic field is smaller than the roll and yaw components (X- and Z-components, respectively, in body frame) during mission mode, and the latter have approximately a sinusoidal time dependence with a period equal to the orbital period. As a result, a spacecraft pitch dipole results primarily in a torque in the roll/yaw plane, and a dipole in the roll/yaw plane results primarily in a torque of approximately the same magnitude about the pitch axis.

The ADGEN system contains a differential corrector which utilizes the attitude solutions from the attitude history file, Sun and spacecraft ephemeris, and electromagnet and scanwheel speed data to obtain estimates of the spacecraft residual dipole. The system data flow is shown in Figure 8-6.

8.3.1 Analytical Considerations

8.3.1.1 Ephemeris Generation

The spacecraft ephemeris is obtained from a definitive orbit tape in the EPHEM format. An internal orbit generator is required only for system testing. The inertial position of the Sun is either set as a constant or obtained from a Goddard Trajectory Determination System (GTDS) Solar-Lunar-Planetary file. The best available magnetic field model should be used to generate the Earth's magnetic field.

8.3.1.2 Coordinate Systems and Transformations

The orbit normal vector $\hat{n}$ is determined from the position $(\bar{p})$ and velocity $(\bar{v})$ of the spacecraft in geocentric inertial coordinates, i.e.,

$$\hat{Y} = -\hat{n} = -\frac{\hat{p} \times \hat{v}}{|\hat{p} \times \hat{v}|} \tag{8-9}$$

$$\hat{Z} = -\hat{p}$$

The orbital coordinate system is a spacecraft-centered coordinate system. The Y (pitch)-axis is the negative orbit normal; the Z (yaw)-axis is the nadir

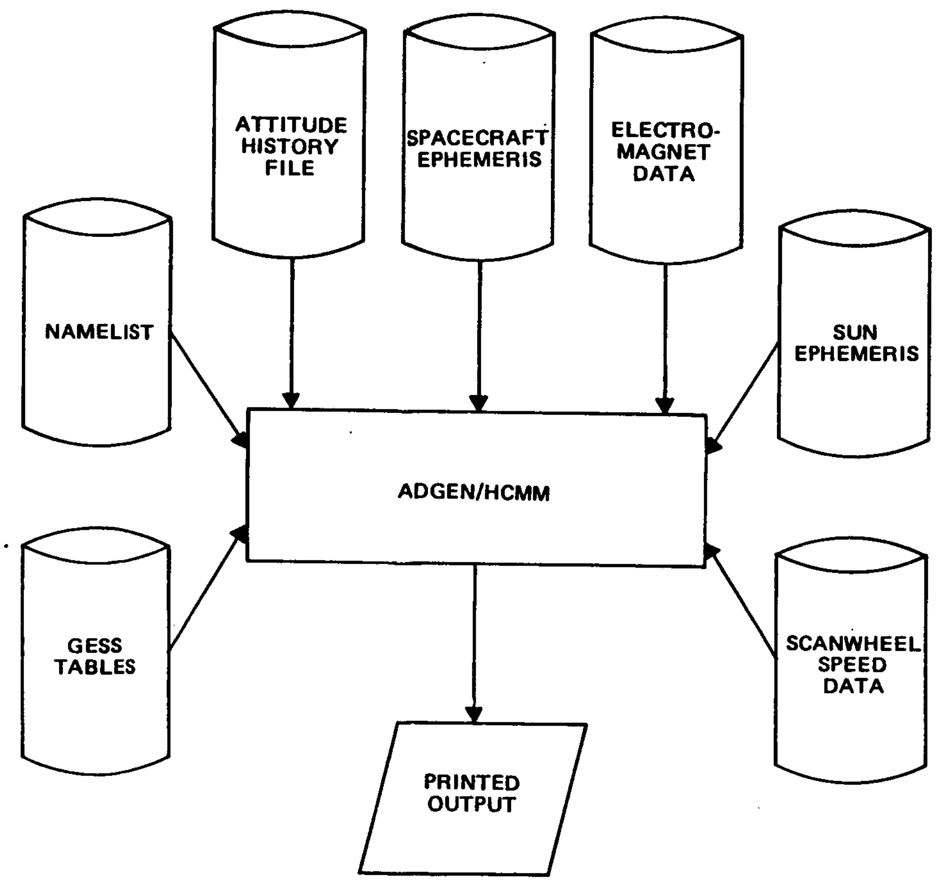


Figure 8-6. ADGEN/HCMM Data Flow

or local vertical, positive toward the geocenter; the X (roll)-axis completes a righthanded system, i.e.,

$$\hat{X} = \hat{Y} \times \hat{Z} \tag{8-10}$$

The body principal coordinate system is centered at the spacecraft center of mass. The coordinate axes of the body principal coordinate system are the three principal axes of the spacecraft inertia ellipsoid, i.e., the moment of inertia tensor is diagonal in this coordinate system. The Z-axis is that coordinate axis nearest the spacecraft Z-axis, and the X-axis is the remaining coordinate axis nearer the spacecraft X-axis.

The transformation matrix [C] from orbital to spacecraft coordinates is expressed as an Euler 2-1-3 rotation (pitch, roll, and yaw, respectively).

$$[C] = \begin{bmatrix} \cos p \cos y + \sin p \sin r \sin y & \cos r \sin y & -\sin p \cos y + \cos p \sin r \sin y \\ -\cos p \sin y + \sin p \sin r \cos y & \cos r \cos y & \sin p \sin y + \cos p \sin r \cos y \\ \sin p \cos r & -\sin r & \cos p \cos r \end{bmatrix} \tag{8-11}$$

The transformation matrix [G] from inertial to orbital coordinates is given by

$$[G] = \begin{bmatrix} \hat{X}^T \\ \hline \hat{Y}^T \\ \hline \hat{Z}^T \end{bmatrix} \tag{8-12}$$

where $\hat{X}$, $\hat{Y}$, and $\hat{Z}$ are given in Equations (8-9) and (8-10).

The transformation matrix [H] from inertial to body principal coordinates is

$$[H] = [B]\,[C]\,[G] \tag{8-13}$$

where [B] is the transformation from spacecraft to body principal coordinates.

8.3.1.3 Spacecraft Dynamics and Integration

The total spacecraft angular momentum is

$$\bar{H} = I\bar{W} + \bar{H}' \tag{8-14}$$

where $\bar{H}'$ = wheel angular momentum vector

I = moment of inertia tensor

$\bar{W}$ = spacecraft angular velocity vector

$\bar{H}$ = total spacecraft angular momentum

The external torques consist of a solar radiation pressure torque, magnetic torques, and a gravity-gradient torque expressed in the spacecraft body coordinates. These external torques are discussed in Section 7.3.1. The equations of motion are

$$\dot{\bar{H}} = \bar{N} - \bar{W} \times \bar{H} = \bar{N} - I^{-1} (\bar{H} - \bar{H}') \times \bar{H} \tag{8-15a}$$

and

$$\dot{\bar{E}} = 1/2 \; [\Omega] \; \bar{E} \tag{8-15b}$$

where $\bar{N}$ = total external torque vector

$\bar{E}$ = vector of four Euler symmetric parameters

and

$$[\Omega] = \begin{pmatrix} 0 & W_Z & -W_Y & W_X \\ -W_Z & 0 & W_X & W_Y \\ W_Y & -W_X & 0 & W_Z \\ -W_X & -W_Y & -W_Z & 0 \end{pmatrix} \tag{8-16}$$

This vector consisting of the four Euler symmetric parameters and the spacecraft angular momentum vector is referred to as the attitude state vector, to be distinguished from the differential corrector state vector. These equations of motion are integrated using an Adams-Bashforth variable step-size predictor-corrector, which is discussed in Section 7.3.2. The vectors represented in the equations of motion are expressed in body principal coordinates.

Initial estimates of the spacecraft angular velocity vector and attitude are required to begin integration of the equations of motion. To obtain these estimates, the attitude solutions generated by the HCMM ADS corresponding to the start time of the pass are read from the attitude history file. These solutions are in the form of pitch (p), roll (r), and yaw (y) angles. The initial estimates of the pitch, roll, and yaw rates are input parameters.

The spacecraft angular velocity vector in spacecraft coordinates, $\bar{W}_B$, is computed from these attitude solutions by

$$\bar{W}_B = \begin{pmatrix} \sin y \cos r \, (\dot{p} - \Omega_o) + \cos y \, \dot{r} \\ \cos r \cos y \, (\dot{p} - \Omega_o) - \sin y \, \dot{r} \\ \dot{y} - \sin r \, (\dot{p} - \Omega_o) \end{pmatrix} \tag{8-17}$$

where Ω_o is the orbital rate.

The angular velocity vector is then transformed into body principal coordinates by

$$\bar{W}_p = [B] \, \bar{W}_B \tag{8-18}$$

The pitch, roll, and yaw angles define the orbital to body transformation [C] and hence the attitude matrix [H] (Equation (8-13)), because the spacecraft ephemeris is known. The Euler symmetric parameters are related to the elements of the attitude matrix by

$$4 E_1^2 = 1 + H_{11} - H_{22} - H_{33} \tag{8-19a}$$

$$4 E_2^2 = 1 - H_{11} + H_{22} - H_{33} \tag{8-19b}$$

$$4 E_3^2 = 1 - H_{11} - H_{22} + H_{33} \tag{8-19c}$$

$$4 E_4^2 = 1 + H_{11} + H_{22} + H_{33} \tag{8-19d}$$

$$4 E_1 E_2 = H_{12} + H_{21} \tag{8-20a}$$

$$4 E_1 E_3 = H_{31} + H_{13} \tag{8-20b}$$

$$4 E_1 E_4 = H_{23} - H_{32} \tag{8-20c}$$

$$4 E_2 E_3 = H_{23} + H_{32} \tag{8-20d}$$

$$4 E_2 E_4 = H_{31} - H_{13} \tag{8-20e}$$

$$4 E_3 E_4 = H_{12} - H_{21} \tag{8-20f}$$

Equations (8-19) are searched in order d, a, b, c until one is found for which the right side is greater than 10^{-5}. The positive square root is taken to find one Euler parameter, and the other three are found by solving three equations of Equation (8-20). Note that the matrix components do not uniquely determine the overall sign of the Euler symmetric parameters.

8.3.1.4 Differential Corrector

The ADGEN/HCMM differential corrector is designed to estimate the spacecraft residual magnetic dipole and the pitch, roll, and yaw rates, based on attitude solutions from the HCMM ADS, ephemeris data, and data from the spacecraft control hardware. The differential corrector is a Gauss-Newton filter, and a detailed mathematical description of this filter is given in Section 4.3.3. Thus, the emphasis in this section is on the application of the filter to the specific task of residual magnetic dipole determination.

The differential corrector state vector (or solve-for variable vector), $\bar{x}$, is the 10-element vector

$$\bar{x} = \left(d_1, d_2, d_3, W_1^0, W_2^0, W_3^0, E_1^0, E_2^0, E_3^0, E_4^0\right)^T \tag{8-21}$$

where $\bar{d} = (d_1, d_2, d_3)^T$ is the residual magnetic dipole in spacecraft coordinates, $\bar{W}^0 = \left(W_1^0, W_2^0, W_3^0\right)^T$ is the initial value of the spacecraft angular momentum vector in spacecraft coordinates, and $\bar{E}^0 = \left(E_1^0, E_2^0, E_3^0, E_4^0\right)^T$ is the initial value of the Euler symmetric parameters.

The observation vector $\bar{y}$ at time t is the vector of the four Euler symmetric parameters

$$\bar{y}(t) = (E_1(t), E_2(t), E_3(t), E_4(t))^T \tag{8-22}$$

The function $\bar{f}(\bar{x}, \bar{m})$, from which the observation model vector $\bar{z}$ is derived, is not expressed in closed form. Rather, it is evaluated dynamically by integrating the equations of motion (Equation (8-14)). The vector of measurements required to integrate the equations of motion is

$$\bar{m} = (s_1, C_1, C_2, C_3) \tag{8-23}$$

where s is the scanwheel speed and $\bar{C} = (C_1, C_2, C_3)$ is the vector of the three electromagnet coil readings. The scanwheel determines the wheel angular momentum vector, while the electromagnet coil readings determine the magnetic torque on the spacecraft.

Because the observation model vector $\bar{f}(\bar{x}, \bar{m})$ is evaluated dynamically, special considerations must be given to calculating its partial derivative with respect to the state vector elements. Two different types of dynamic solve-for variables are distinguished in ADGEN. The first are initial values of the state parameters, and the second are parameters affecting the evolution of the system from the initial state. Examples of the first type are initial values of the Euler symmetric parameters, and initial values of the components of the spacecraft body angular velocity. Examples of the second type are the residual magnetic dipole moments.

Analytic partial derivatives of the observation model vector with respect to the variables of the first type are available in ADGEN. These can be used with any attitude state vector representation, but are only approximate, because they are based on closed-form expressions valid only for constant spacecraft body rates. However, they are still useful if the body rates are not constant, because the differential corrector does not need precise values of the partial derivatives, in general.

An alternative method of computing partial derivatives in ADGEN is by numerical integration. This method is available for all dynamic solve-for parameters, and must be used if any parameters of the second type are to be estimated. These partial derivatives are, in principle, more precise than the analytic partials, but are currently available only if the attitude state equations of motion are those given by Equation (8-15).

To simplify the discussion of the numerical partial derivative calculations, let the attitude state vector be represented by $y_i(\alpha_\lambda, t)$ where $i = 1, 2, \ldots, k$ ($k = 7$ in this case), and let α_λ, $\lambda = 1, 2, \ldots, n$ be the dynamic solve-for parameters. We do not have closed-form solutions for the y_i's, of course, but we have the equations of motion $\dot{y}_i = f_i(y_i, \alpha_\lambda, t)$. We differentiate these equations with respect to α_λ, remembering that f_i depends on the solve parameters both explicitly and implicitly, through the y_i dependence. Thus,

$$\frac{\partial \dot{y}_i}{\partial \alpha_\lambda} = \frac{\partial f_i}{\partial y_i}\frac{\partial y_i}{\partial \alpha_\lambda} + \left.\frac{\partial f_i}{\partial \alpha_\lambda}\right|_{\text{explicit}} = \frac{d}{dt}\frac{\partial y_i}{\partial \alpha_\lambda} \tag{8-24}$$

The last step follows from the time independence of the solve-for parameters. With the notation

$$\begin{aligned} P_{i\lambda} &\equiv \frac{\partial y_i}{\partial \alpha_\lambda} \\ M_{ij} &\equiv \frac{\partial f_i}{\partial y_i} \\ Q_{i\lambda} &\equiv \left.\frac{\partial f_i}{\partial \alpha_\lambda}\right|_{\text{explicit}} \end{aligned} \tag{8-25}$$

we see that the partial derivatives are the solution of the linear coupled differential equations

$$\dot{P} = MP + Q \tag{8-26}$$

where P , M , and Q are matrices with elements given by Equation (8-25). The partials can thus be computed numerically if M , Q , and initial values of P are known.

8.3.2 ADGEN/HCMM Input

Input for ADGEN/HCMM consists of attitude solutions for each pass from the attitude history file, spacecraft control system data from the electromagnet and scanwheel rate data sets, spacecraft and Sun ephemeris, program control parameters from the GESS displays, and NAMELIST parameters.

The attitude solutions obtained from the attitude history file consist of the pitch, roll, and yaw angles. The spacecraft ephemeris provides the spacecraft position and velocity vectors in inertial coordinates, while the Sun unit vector in inertial coordinates is obtained from Sun ephemeris. The scanwheel speed (rpm) and electromagnet strength (pole-centimeters) are obtained from the scanwheel and electromagnet data sets, respectively.

The dynamics NAMELIST parameters include

- Components of spacecraft angular velocity along spacecraft body principal axes for each data segment
- Inertia tensor in body geometric coordinates
- Start time for each data segment
- Stop time for each data segment
- Integration step size
- Flag to determine how often torques and wheel rates are updated

- Spacecraft ephemeris option flag
- Epoch of orbital elements
- Spacecraft orbital elements
- Flag for gravity-gradient torque
- Source of magnetic dipole information
- Number of onboard magnetic dipoles
- Mounting angles of magnetic dipoles
- Initial estimate of magnetic dipole moment
- Number of momentum wheels
- Mounting angles of momentum wheels
- Moments of inertia of momentum wheels
- Diagnostic output level flags
- FORTRAN data set reference numbers for input and output
- Solar radiation pressure torque parameters

Differential corrector NAMELIST parameters include

- Flags indicating solve-for parameters
- A priori covariance flag
- A priori covariances of state variables
- Convergence tolerances for each solve-for parameter
- Maximum number of allowed iterations
- Edit flags for observations
- Standard deviations of observation types
- Tolerances for dynamic editing of observation types

- Sigma rejection factor for dynamic editing
- Weighted root-mean-square (rms) used for dynamic editing on first iteration
- Initial time of span for observations processing
- Final time of span for observations processing
- Control flags for various reports
- Level flags for diagnostic output
- Maximum number of lines per page for printed output
- FORTRAN data set reference numbers for input and output

8.3.3 ADGEN/HCMM Functional Description

The ADGEN/HCMM System is subdivided into the following four functional modules:

- Initialization module
- Differential corrector module
- Dynamics module
- Output module

Functional descriptions of these modules follow.

8.3.3.1 Initialization Module

The initialization module initiates the system and prepares it to process the data from the attitude history file and spacecraft contol data sets. These functions include reading the dynamics and differential corrector NAMELISTs, converting selected input parameters to the correct format and units, transforming the input attitude solution, initializing parameters and arrays, and building the observations working file.

8.3.3.2 Differential Corrector Module

The differential corrector reads an observation from the observations working file and calls the dynamics module to integrate the equations of motion forward to the observation time, using the initial attitude state vector. The observation model is used to determine a computed value of pitch, roll, and yaw based on the integrated attitude and a residual at the observation time. Two data editing tests are performed. An absolute tolerance test and a sigma rejection test are performed. The data point is rejected if it fails either test. If both tests are passed, the partial derivations of the observation with respect to the solve-for variables are computed, and the normal matrix is updated. After all observations from the observations working file have been processed, the normal matrix is inverted and a new estimate of the state is computed. This process is repeated until the maximum specified number of iterations is performed or the calculation converges. The convergence criterion is that the change in each solve-for variable is less than the tolerance value specified for that variable.

8.3.3.3 Dynamics Module

The dynamics module determines the environmental torques, the angular momentum from the scanner wheel, and the derivatives of the attitude state vector. In addition, it solves the equations of motion and propagates the attitude state vector to the observation time. The electromagnet coil data is combined with the Earth's magnetic field vector generated by the spherical harmonic expansion and the estimate of the residual dipole to compute the magnetic torque. The scanwheel speed data is used to evaluate the angular velocity vector of the spacecraft. The attitude state vector, which is expressed in terms of Euler symmetric parameters and angular velocity components in body principal coordinates, is then propagated using the Adams-Bashforth predictor-corrector in a specified step size to the observation time.

8.3.3.4 Output Module

The output module generates all hardcopy printout and displays produced by the ADGEN/HCMM system. This output consists of the residual report, the end-of-iteration report, residual printer plots, and error messages.

8.3.4 ADGEN/HCMM Output

At the end of each iteration of the differential corrector, the output to hardcopy printout consists of

- Start time of processing interval
- Stop time of processing interval
- Estimate of electromagnetic dipole biases and the standard deviation
- Mean of observation residuals by observation type
- Standard deviation of observation residuals by observation type
- Covariance/correlation matrix
- Observation residuals

8.3.5 ADGEN/HCMM Displays

ADGEN/HCMM can optionally generate many displays for diagnostic purposes. Others are always generated in nominal processing, and are described in the following subsections. Pertinent output data are produced in hardcopy; GESS displays need not be printed.

8.3.5.1 NAMELIST Displays

The NAMELIST displays will contain all parameters input through NAMELIST which are listed in Section 5.3.2.

8.3.5.2 Residual Report Displays

The residual report displays will contain

- Observation time
- Observation type
- Iteration number

- Edit flag
- Observed value
- Computed value
- Residual--observed minus computed value
- Observation weight
- Observation number

The operator may reject data from this display.

8.3.5.3 Solve Parameter Results Display

The solve parameter results display will contain

- Parameter name
- Current value of parameter
- Parameter standard deviation
- Parameter change from previous iteration
- Parameter change from a priori value
- A priori value of parameter
- Convergence indicator
- Iteration number
- Data start and stop times
- Combined weighted rms

8.3.4.4 Iteration Summary Display

The iteration summary display will contain

- Parameter name
- Value of solve parameter for previous six iterations
- Convergence indicator
- Iteration number
- Data start and stop times

8.3.5.5 Observation Summary Display

The observation summary display will contain

- Iteration number
- Observation data type
- Number of observations for each data type
- Number of observations accepted for each data type
- Weighted rms of each data type
- Mean residual of each data type
- Standard deviation of each data type
- Data start and stop times
- Combined weighted rms

8.3.5.6 Covariance/Correlation Matrix Display

The covariance/correlation matrix display will contain

- Parameter name
- Combined weighted rms
- Iteration number
- Data start and stop times
- Covariance/correlation matrix

8.3.5.7 Residual Plot Display

The residual plot display will contain

- Observation type
- Plot of observation residuals versus time

The format of these displays will be provided in a modification to the specifications.

APPENDIX A - EVALUATION OF THE MATRIX OF PARTIAL DERIVATIVES

The definitions of the state vector $\bar{x} = (p, p_1, p_2, r, r_1, r_2, y, y_1, y_2, b_1, b_2, b_3)^T$ and the observation model vector $\bar{f}(\bar{x}, \bar{m})$ are given in Equations (4-109) and (4-114), respectively. The partial derivatives are (T=time)

$$\frac{\partial f_1}{\partial x_1} = \text{cycr}/[\text{spcy} - \text{cpsrsy})^2 + (\text{cpcr})^2] \tag{A-1}$$

$$\frac{\partial f_1}{\partial x_2} = \frac{\partial f_1}{\partial x_1} \cdot T \tag{A-2}$$

$$\frac{\partial f_1}{\partial x_3} = \frac{\partial f_1}{\partial x_1} \cdot \frac{T^2}{2} \tag{A-3}$$

$$\frac{\partial f_1}{\partial x_4} = (\text{spsrcy} - \text{cpsy})\ \text{cp}/[\text{spcy} - \text{cpsrsy})^2 + (\text{cpcr})^2] \tag{A-4}$$

$$\frac{\partial f_1}{\partial x_5} = \frac{\partial f_1}{\partial x_4} \cdot T \tag{A-5}$$

$$\frac{\partial f_1}{\partial x_6} = \frac{\partial f_1}{\partial x_4} \cdot \frac{T^2}{2} \tag{A-6}$$

$$\frac{\partial f_1}{\partial x_7} = -\ (\text{spsy} + \text{cpsrcy})\ \text{cpcr}/[(-\text{spcy} + \text{cpsrsy})^2 + (\text{cpcr})^2] \tag{A-7}$$

$$\frac{\partial f_1}{\partial x_8} = \frac{\partial f_1}{\partial x_7} \cdot T \qquad \text{(A-8)}$$

$$\frac{\partial f_1}{\partial x_9} = \frac{\partial f_1}{\partial x_7} \cdot \frac{T}{2} \qquad \text{(A-9)}$$

$$\frac{\partial f_1}{\partial x_{10}} = \frac{\partial f_1}{\partial x_{11}} = \frac{\partial f_1}{\partial x_{12}} = 0 \qquad \text{(A-10)}$$

$$\frac{\partial f_2}{\partial x_1} = (\text{cpsy} - \text{spsrcy})/[1 - (\text{spsy} + \text{cpsrcy})^2]^{1/2} \qquad \text{(A-11)}$$

$$\frac{\partial f_2}{\partial x_2} = \frac{\partial f_2}{\partial x_1} \cdot T \qquad \text{(A-12)}$$

$$\frac{\partial f_2}{\partial x_3} = \frac{\partial f_2}{\partial x_1} \cdot \frac{T^2}{2} \qquad \text{(A-13)}$$

$$\frac{\partial f_2}{\partial x_4} = \text{cpcrcy}/[1 - (\text{spsy} + \text{cpsrcy})^2]^{1/2} \qquad \text{(A-14)}$$

$$\frac{\partial f_2}{\partial x_5} = \frac{\partial f_2}{\partial x_4} \cdot T \qquad \text{(A-15)}$$

$$\frac{\partial f_2}{\partial x_6} = \frac{\partial f_2}{\partial x_4} \cdot \frac{T^2}{2} \qquad \text{(A-16)}$$

$$\frac{\partial f_2}{\partial x_7} = (\text{spcy} - \text{cpsrsy})/[1 - (\text{spsy} + \text{cpsrcy})^2]^{1/2} \qquad \text{(A-17)}$$

$$\frac{\partial f_2}{\partial x_8} = \frac{\partial f_2}{\partial x_7} \cdot T \qquad \text{(A-18)}$$

$$\frac{\partial f_2}{\partial x_9} = \frac{\partial f_2}{\partial x_7} \cdot \frac{T^2}{2} \qquad \text{(A-19)}$$

$$\frac{\partial f_2}{\partial x_{10}} = \frac{\partial f_2}{\partial x_{11}} = \frac{\partial f_2}{\partial x_{12}} = 0 \qquad \text{(A-20)}$$

$$\frac{\partial f_3}{\partial x_1} = (-\text{spcy} + \text{cpsrsy})\ S_{O1} - (\text{cpcy} + \text{spsrsy})\ S_{O3} \qquad \text{(A-21)}$$

$$\frac{\partial f_3}{\partial x_2} = \frac{\partial f_3}{\partial x_1} \cdot T \qquad \text{(A-22)}$$

$$\frac{\partial f_3}{\partial x_3} = \frac{\partial f_3}{\partial x_1} \cdot \frac{T^2}{2} \qquad \text{(A-23)}$$

$$\frac{\partial f_3}{\partial x_4} = \text{spcrsy}\ S_{O1} - \text{srsy}\ S_{O2} + \text{cpcrsy}\ S_{O3} \qquad \text{(A-24)}$$

$$\frac{\partial f_3}{\partial x_5} = \frac{\partial f_3}{\partial x_4} \cdot T \qquad \text{(A-25)}$$

$$\frac{\partial f_3}{\partial x_6} = \frac{\partial f_3}{\partial x_4} \cdot \frac{T^2}{2} \qquad \text{(A-26)}$$

$$\frac{\partial f_3}{\partial x_7} = (-\text{cpsy} + \text{spsrcy})\ S_{O1} + \text{crcy}\ S_{O2} + (\text{spsy} + \text{cpsrcy})\ S_{O3} \qquad \text{(A-27)}$$

$$\frac{\partial f_3}{\partial x_8} = \frac{\partial f_3}{\partial x_7} \cdot T \qquad \text{(A-28)}$$

$$\frac{\partial f_3}{\partial x_9} = \frac{\partial f_3}{\partial x_7} \cdot \frac{T^2}{2} \qquad \text{(A-29}$$

$$\frac{\partial f_3}{\partial x_{10}} = \frac{\partial f_3}{\partial x_{11}} = \frac{\partial f_3}{\partial x_{12}} = 0 \qquad \text{(A-30)}$$

$$\frac{\partial f_4}{\partial x_1} = (\text{spsy} + \text{cpsrcy})\ S_{O1} + (\text{cpsy} - \text{spsrcy})\ S_{O3} \qquad \text{(A-31)}$$

$$\frac{\partial f_4}{\partial x_2} = \frac{\partial f_4}{\partial x_1} \cdot T \qquad \text{(A-32)}$$

$$\frac{\partial f_4}{\partial x_3} = \frac{\partial f_4}{\partial x_1} \cdot \frac{T^2}{2} \qquad \text{(A-33)}$$

$$\frac{\partial f_4}{\partial x_4} = \text{spcrcy}\ S_{O1} - \text{srcy}\ S_{O2} + \text{cpcrcy}\ S_{O3} \qquad \text{(A-34)}$$

$$\frac{\partial f_5}{\partial x_4} = - \text{spsr } S_{O1} - \text{cr } S_{O2} - \text{cpsr } S_{O3} \tag{A-44}$$

$$\frac{\partial f_5}{\partial x_5} = \frac{\partial f_5}{\partial x_4} T \tag{A-45}$$

$$\frac{\partial f_5}{\partial x_6} = \frac{\partial f_5}{\partial x_4} \cdot \frac{T^2}{2} \tag{A-46}$$

$$\frac{\partial f_5}{\partial x_7} = \frac{\partial f_5}{\partial x_8} = \frac{\partial f_5}{\partial x_9} = \frac{\partial f_5}{\partial x_{10}} = \frac{\partial f_5}{\partial x_{11}} = \frac{\partial f_5}{\partial x_{12}} = 0 \tag{A-47}$$

The partial derivatives of f_6, f_7, and f_8 can be generated by substituting $\bar{M}_O$ for $\hat{S}_O$ in the partial derivatives of f_3, f_4, and f_5 (Equations (A-21) through (A-35)), respectively, except:

$$\frac{\partial f_6}{\partial x_{10}} = 1 \tag{A-48}$$

$$\frac{\partial f_7}{\partial x_{11}} = 1 \tag{A-49}$$

$$\frac{\partial f_8}{\partial x_{12}} = 1 \tag{A-50}$$

$$\frac{\partial f_4}{\partial x_5} = \frac{\partial f_4}{\partial x_4} \cdot T \tag{A-35}$$

$$\frac{\partial f_4}{\partial x_6} = \frac{\partial f_4}{\partial x_4} \cdot \frac{T^2}{2} \tag{A-36}$$

$$\frac{\partial f_4}{\partial x_7} = -\ (\text{cpcy} + \text{spsrsy})\ S_{O1} - \text{crsy}\ S_{O2} + (\text{spcy} - \text{cpsrsy})\ S_{O3} \tag{A-37}$$

$$\frac{\partial f_4}{\partial x_8} = \frac{\partial f_4}{\partial x_7} \cdot T \tag{A-38}$$

$$\frac{\partial f_4}{\partial x_9} = \frac{\partial f_4}{\partial x_7} \cdot \frac{T^2}{2} \tag{A-39}$$

$$\frac{\partial f_4}{\partial x_{10}} = \frac{\partial f_4}{\partial x_{11}} = \frac{\partial f_4}{\partial x_{12}} = 0 \tag{A-40}$$

$$\frac{\partial f_5}{\partial x_1} = \text{cpcr}\ S_{O1} - \text{spcr}\ S_{O3} \tag{A-41}$$

$$\frac{\partial f_5}{\partial x_2} = \frac{\partial f_5}{\partial x_1} \cdot T \tag{A-42}$$

$$\frac{\partial f_5}{\partial x_3} = \frac{\partial f_5}{\partial x_1} \cdot \frac{T^2}{2} \tag{A-43}$$

OPTIONAL FORM NO. 10
JULY 1973 EDITION
GSA FPMR (41 CFR) 101-11.6

UNITED STATES GOVERNMENT

Memorandum

TO : Mr. Edward J. Prokop
Control Center Systems Branch

DATE: November 22, 1976

FROM : Mission Support and Analysis Branch

SUBJECT: Real Time Data Transmission from MSOCC to Attitude Operations for AEM-A (HCMM)

Attachment 1 is our proposed format for the real time telemetry record being sent from MSOCC to ADCS in support of the AEM-A mission. The record is 312 bytes in length including 24 bytes of header information. Each record will consist of the header and four minor frames. Attachment 2 is a description of our proposed header frame and a description of the GMT timing information to be attached to each minor frame.

If there are any questions concerning our proposed format, please contact me.

Richard S. Nankervis

Richard S. Nankervis
Attitude Determination and Control Section

Attachment - 2

cc: Mr. R. T. Groves, Code 580, w/attachs.
Mr. R. E. Coady, Code 581, w/attachs.
Mr. R. D. Werking, Code 581.2, w/attachs.
Mr. R. Sanford, Code 513, w/attachs.
Mr. G. F. Meyers, Code 581.2, w/attachs.
Mr. G. D. Repass, Code 581.2, w/attachs.
Mr. S. Hotovy, CSC, w/attachs.

581-1088M:RSN:pwd

APPENDIX B - MEMORANDA ON TELEMETRY AND DEFINITIVE ATTITUDE DATA TRANSMISSION

This appendix consists of two memoranda written by the ACA detailing his understanding of telemetry and definitive attitude data transmission. The first is a memorandum to MSOCC concerning telemetry data transmission, and the second is a memorandum to IPD concerning both telemetry data and definitive data transmission.

OPTIONAL FORM NO. 10
JULY 1973 EDITION
GSA FPMR (41 CFR) 101-11.6

UNITED STATES GOVERNMENT

Memorandum

: Messrs. P. L. McKowan and W. M. Watt
Information Processing Division

DATE: November 23, 1976

FROM : Mission Support and Analysis Branch
Mission Support Computing and Analysis Division

SUBJECT: Telemetry Data Transmission from IPD to ADCS and Definitive Attitude Data Transmission from ADCS to IPD for AEM-A (HCMM)

I. Telemetry Data from IPD to ADCS

Attachment 1 is the format we propose be used for transmission of telemetry data from IPD to ADCS in support of the AEM-A mission. The record is 3492 bytes with a 24 byte header. Each record consists of the header and 56 minor frames.

It is agreed that data from IPD will be defined on a pass-by-pass basis and as such the change in station name in the header will constitute a new data interval. Data within each data interval will be chronologically increasing in time. It is understood that the procedure by which each data record is created will result in minor frame "0", if it exists, being the first frame of data in each record and that fill data (zeros) will be inserted in place of any minor frame in a sequence which is noisy or missing. A typical frame is shown in Attachment 2. It is also understood that a change in the spacecraft data mode (telemetry word 48) will result in the existing minor frame and all following minor frames in the record being filled with zeros and a new record being started. Finally we understand that no major frame or complete record will contain only fill data.

II. Attitude Data from ADCS to IPD

Attachment 3 is our proposed format for the definitive attitude records being sent to IPD from ADCS. Each record is 3492 bytes. We will process the telemetry data on a pass-by-pass basis and definitive results will be transmitted to IPD on a pass-by-pass basis. Definitive results for a pass will consist of one header record and as many data records as required. After receipt of telemetry data from IPD and corresponding definitive orbit tapes we will generate definitive attitude data and transmit these results to IPD. We will generate and transmit definitive attitude upon receipt of all necessary data and thus transmission of passes will not necessarily be in chronological order. Attempts will be made to smooth results and as such to be able to interpolate across times when telemetry data does not exist, however, there may be times in which interpolation during a pass will not be possible. If this happens, the only indication of such an occurrence will be a jump in the time attached to the definitive attitude results.

Subject: Telemetry Data Transmission from IPD to ADCS and Definitive Attitude Data Transmission from ADCS to IPD for AEM-A (HCMM)

The header record and the data record will consist of the header frame and data as specified in Attachment 3. Each definitive data record will contain 144 data blocks which, at a nominal output frequency of 10 seconds per block, will contain 24 minutes of data. This means that nominally each pass of data will consist of only one data record and will contain about half fill data. Attachment 4 is a memorandum pertaining to telemetry interfaces between IPD and ADCS.

If there are any questions pertaining to our proposed formats, please contact me.

Richard S. Nankervis

Richard S. Nankervis
Attitude Determination and Control Section

Attachments - 4

cc: Mr. R. T. Groves, Code 580, w/attachs.
Mr. R. E. Coady, Code 581, w/attachs.
Mr. R. D. Werking, Code 581.2, w/attachs.
Mr. R. Sanford, Code 513, w/attachs.
Mr. G. F. Meyers, Code 581.2, w/attachs.
Mr. G. D. Repass, Code 581.2, w/attachs.
Mr. S. Hotovy, CSC, w/attachs.

581-1089M:RSN:pwd

APPENDIX C - DATA REDUCTION NOTES

This appendix contains information which will aid in the conversion of attitude data from the telemetry stream format to appropriate engineering units. The appendix consists of a bit-by-bit description of the base module telemetry stream, a description of Pulse Code Modulation (PCM) digital data, a list of PCM analog data conversion factors and conversion tables, and Sun sensor angle conversion tables.

SECTION 3

QUICK-LOOK TELEMETRY LISTS

This section contains lists of all PCM-telemetered functions and parameters referenced to their locations in Launch, HCMM Orbital, and SAGE Orbital formats Tables 3-1 through 3-3.

These lists have been prepared to assist in formatting software and as quick-look references. The following comments, based on the Boeing Company numerical designations for the Base Module, apply:

- Word—The 8-bit word position in the 128-word minor frame, numbered from 1 to 128.
- Frame—The minor frame position in a 1 to 32 numbering position. Launch format, HCMM format, and SAGE formats are eight minor frames, and Memory Dump Format is 32 minor frames per major frame. (See Table 2.2.)
- Bit—Bit position in an 8-bit word, where "Bit 1" is the MSB and "Bit 8" is the LSB.
- ID—An alphanumeric designator that is unique to each function/parameter. For user convenience, this column is located at both sides of the tables.
- Type—Identifies the type of input to the encoder. (See list of abbreviations in this section.)
- Parameter Range—A generalized description of the probable range/function for each function. Actual scaling and conversion factors for analog parameters are presented in Section 9 and 10.

Because of its simplicity, information on the Memory Dump Format is presented in Section 8.

APPENDIX C - DATA REDUCTION NOTES

This appendix contains information which will aid in the conversion of attitude data from the telemetry stream format to appropriate engineering units. The appendix consists of a bit-by-bit description of the base module telemetry stream, a description of Pulse Code Modulation (PCM) digital data, a list of PCM analog data conversion factors and conversion tables, and Sun sensor angle conversion tables.

Table 3-1 (Continued)

ID	Measurement Name	Subsys.	Type	Parameter Range	Launch			HCMM Orbital			Sage Orbital			ID
					Word	Frame	Bits	Word	Frame	Bits	Word	Frame	Bits	
A26	5-volt supply	ACS	HLA	0 to +5V	N/A			93	4	All	110	4	All	A26
A41	10-volt supply	ACS	HLA	+6.2 to 11.2V	88	4	All	125	All	All	77	8	All	A41
A21	Magnetic field rate, (roll)	ACS	HLA	±0.012 Gauss/sec	N/A			112	All	All	78	5	All	A21
A22	Magnetic field rate, (pitch)	ACS	HLA	±0.012 Gauss/sec	N/A			116	All	All	78	6	All	A22
A23	Magnetic field rate, (yaw)	ACS	HLA	±0.012 Gauss/sec	N/A			117	All	All	78	7	All	A23
A20	Spin accelerometer	ACS	HLA	0 to 200 rpm	80	4	All	114	All	All	79	7	All	A20
A18	Sun sensor ID	ACS	BL	0 to 5	N/A			58	All	6-8	78	3	6-8	A18
A16	Sun sensor, reticle A	ACS	BL	±64°	N/A			59	All	All	79	4	All	A16
A17	Sun sensor, reticle B	ACS	BL	±64°	N/A			60	All	All	79	5	All	A17
A19	ACS acq./on orbit mode	ACS	BL	1 = Acq. mode	N/A			58	All	5	78	3	5	A19
A42	Pitch 0/±90 (SAGE)	ACS	BL	1 = ±90	N/A			58	All	4	N/A			A42
A32	Magnetometer on/off	ACS	BL	1 = On	N/A			58	All	3	78	3	3	A32
A34	Magnetometer in/out of loop	ACS	BL	1 = Out	N/A			58	All	2	78	3	2	A34
A35	Automatic reacq. in/out	ACS	BL	1 = In	N/A			58	All	1	78	3	1	A35
F1	Sun sensor on/off	ACS	BL	1 = On	104	8	4	77	8	4	109	8	4	F1
F2	Spin accelerometer on/off	ACS	BL	1 = On	104	8	2	77	8	5	109	8	5	F2
F26	Stage 4 TM on/off	LV	BL	1 = On	77	4	8	77	4	8	109	4	8	F26
S7	Stage 4 pressure	LV	HLA	0 to 750 PSI	1 + 8n (n = 0–15)	All	All	N/A			N/A			S7
S8	Stage 4 Z acceleration	LV	HLA	-1 to +30 G's	2 + 8n	All	All	N/A			N/A			S8
S9	Stage 4 X vibration	LV	HLA	±0.5 G	3 + 8n	All	All	N/A			N/A			S9
S10	Stage 4 Y vibration	LV	HLA	±0.8 G	4 + 8n	All	All	N/A			N/A			S10
S11	Stage 4 Z vibration	LV	HLA	±0.5 G	5 + 8n	All	All	N/A			N/A			S11
S17	Solar paddle 1 deploy	Deploy	BL	0 = Deployed	80	8	7	74	All	7	78	4	7	S17
S18	Solar paddle 2 deploy	Deploy	BL	0 = Deployed	80	8	8	74	All	8	78	4	8	S18
S19	Antenna boom 1 deploy	Deploy	BL	0 = Deployed	80	8	5	N/A			78	4	5	S19
S20	Antenna boom 2 deploy	Deploy	BL	0 = Deployed	80	8	6	N/A			78	4	6	S20

Table 3-1
Base Module Telemetry List

ID	Measurement Name	Subsys.	Type	Parameter Range	Launch Word	Launch Frame	Launch Bits	HCMM Orbital Word	HCMM Orbital Frame	HCMM Orbital Bits	SAGE Orbital Word	SAGE Orbital Frame	SAGE Orbital Bits	ID
A1	Magnetometer roll (X)	ACS	HLA	±0.6 Gauss	N/A			4, 36, 68, 100	All	All	98	All	All	A1
A2	Magnetometer pitch (Y)	ACS	HLA	±0.6 Gauss	N/A			5, 37, 69, 101	All	All	113	All	All	A2
A3	Magnetometer yaw (Z)	ACS	HLA	±0.6 Gauss	N/A			6, 38, 70, 102	All	All	114	All	All	A3
A5	Electromagnet bias X (roll)	ACS	HLA	±10,000 pole-cm	N/A			17	All	All	77	1	All	A5
A6	Electromagnet bias Y (pitch)	ACS	HLA	±10,000 pole-cm	N/A			18	All	All	77	2	All	A6
A7	Electromagnet bias Z (yaw)	ACS	HLA	±10,000 pole-cm	N/A			20	All	All	77	3	All	A7
A8	Electromagnet moment (roll)	ACS	HLA	±10,000 pole-cm	N/A			21	All	All	77	4	All	A8
A9	Electromagnet moment (pitch)	ACS	HLA	±10,000 pole-cm	N/A			33	All	All	77	5	All	A9
A10	Electromagnet moment (yaw)	ACS	HLA	±10,000 pole-cm	N/A			34	All	All	77	6	All	A10
A27	Roll error (fine)	ACS	HLA	±5°	N/A			11	All	All	80	5	All	A27
A11	Roll error (coarse)	ACS	HLA	±50°	N/A			40	All	All	77	7	All	A11
A28	Pitch error (fine)	ACS	HLA	±2°	N/A			120	All	All	80	6	All	A28
A12	Pitch error (coarse)	ACS	HLA	±50°	N/A			41	All	All	79	1	All	A12
A14	Scan wheel B speed	ACS	HLA	0 to 2200 rpm	N/A			42	All	All	79	3	All	A14
A13	Scan wheel A speed	ACS	HLA	0 to 1000 rpm	N/A			N/A			79	2	All	A13
A15	Roll bias	ACS	HLA	±5°	N/A			43	All	All	N/A			A15
A38	SWA-A H1 duty cycle	ACS	HLA	0 to 50%	N/A			N/A			80	7	All	A38
A39	SWA-A H2 duty cycle	ACS	HLA	0 to 50%	N/A			N/A			80	8	All	A39
A34	SWA-B H1 duty cycle	ACS	HLA	0 to 50%	N/A			119	All	All	80	4	All	A34
A29	SWA-B H2 duty cycle	ACS	HLA	0 to 50%	N/A			122	All	All	78	8	All	A29
A37	Scan wheel A temperature	ACS	HLA	-20 to +60°C	88	2	All	N/A			110	6	All	A37
A25	Scan wheel B temperature	ACS	HLA	-20 to +60°C	112	5	All	93	3	All	110	3	All	A25
A40	SWA-A motor excitation	ACS	HLA	0 to 33 V	N/A			N/A			111	2	All	A40
A30	SWA-B motor excitation	ACS	HLA	0 to 33 V	N/A			123	All	All	110	6	All	A30
A31	power supply temperature	ACS	HLA	-20 to +60°C	111	5	All	124	All	All	110	7	All	A31

Table 3-1 (Continued)

ID	Measurement Name	Subsys.	Type	Parameter Range	Launch			HCMM Orbital			SAGE Orbital			ID
					Word	Frame	Bits	Word	Frame	Bits	Word	Frame	Bits	
F27	Battery over temperature	PWR	BL	1=On	104	2	3	77	2	3	109	2	3	F27
F28	Limiter A bias control	PWR	BL	1=Enable	104	2	1	77	2	1	109	2	1	F28
F29	Limiter B bias control	PWR	BL	1=Enable	104	2	2	77	2	2	109	2	2	F29
F30	Battery thermistor 1 A/B	PWR	BL	1=A	104	2	4	77	2	4	109	2	4	F30
F31	Battery thermistor 2 A/B	PWR	BL	1=A	104	2	5	77	2	5	109	2	5	F31
F32	UV/OI sensor bypass (5-volt)	PWR	BL	1=Off	104	1	1	77	1	1	109	1	1	F32
S1	Equip. panel 1 temp.	THERM	HLA	-20 to +60°C	110	7	All	79	3	All	112	3	All	S1
S2	Equip. panel 2 temp.	THERM	HLA	-20 to +60°C	110	8	All	79	4	All	112	4	All	S2
S3	Equip. panel 3 temp. (Scan wheel A on SAGE)	THERM	HLA	-20 to +60°C	111	1	All	79	5	All	112	5	All	S3
S4	Equip. panel 4 temp.	THERM	HLA	-20 to +60°C	111	2	All	79	6	All	112	6	All	S4
S5	Equip. panel 5 temp.	THERM	HLA	-20 to +60°C	111	3	All	79	7	All	112	7	All	S5
S6	Scan wheel B mount temp.	THERM	HLA	-20 to +60°C	111	4	All	79	8	All	112	8	All	S6
T11	S-band XMTR temp.	CDH	HLA	-20 to +60°C	110	5	All	79	1	All	112	1	All	T11
T20	S-band XMTR RF power	CDH	HLA	0 to 2 watts	96	2	All	93	6	All	93	7	All	T20
T12	VHF XMTR temp.	CDH	HLA	-20 to +60°C	110	6	All	79	2	All	N/A			T12
T21	VHF XMTR RF power	CDH	HLA	0 to 0.25 watt	96	4	All	93	7	All	N/A			T21
T26	XPDR RCVR AGC	CDH	HLA	-50 to -115 dbm	96	7	All	N/A			96	8	All	T26
T23	XPDR RCVR VCO cont. volt	CDH	HLA	±30°	96	5	All	N/A			110	1	All	T23
T22	XPDR XMTR temp.	CDH	HLA	-20 to +60°	96	3	All	N/A			112	2	All	T22
T24	XPDR XMTR RF power	CDH	HLA	0 to watts	96	6	All	N/A			110	2	All	T24
T14	VHF RCVR AGC	CDH	HLA		118	4	All	93	8	All	N/A			T14
T3	A/D CONV CAL level 1	CDH	HLA	0.11 volts	80	1	All	53	All	All	93	4	All	T3
T4	A/D CONV CAL level 2	CDH	HLA	2.50 volts	80	2	All	54	All	All	93	5	All	T4
T5	A/D CONV CAL level 3	CDH	HLA	5.01 volts	80	3	All	56	All	All	93	6	All	T5
T30	Spacecraft address	CDH	INT	001=HCMM, 010=SAGE	48	All	1-3	48	All	1-3	48	All	1-3	T30
T13	PCM mode word	CDH	BL	See 5.3	48	All	4-8	48	All	4-8	48	All	4-8	T13
T15	CMD mode word	CDH	BL	See 5.6	47	All	5-8	47	All	5-8	47	All	5-8	T15

Table 3-1 (Continued)

ID	Measurement Name	Subsys.	Type	Parameter Range	Launch			HCMM Orbital			SAGE Orbital			ID
					Word	Frame	Bits	Word	Frame	Bits	Word	Frame	Bits	
E10	+X Solar paddle temp 1	PWR	HLA	-30 to +100°C	94	2	All	78	7	All	96	4	All	E10
E11	+X Solar paddle temp 2	PWR	LLA	-100 to +100°C	94	3	All	78	8	All	96	5	All	E11
E8	-X Solar paddle temp 1	PWR	LLA	-100 to +100°C	95	8	All	78	5	All	96	2	All	E8
E9	-X Solar paddle temp 2	PWR	HLA	-30 to +100°C	94	1	All	78	6	All	96	3	All	E9
E7	Battery temperature 1	PWR	HLA	-20 to +50°C	95	7	All	78	4	All	96	1	All	E7
E22	Battery temperature 2	PWR	HLA	-20 to +50°C	94	7	All	94	1	All	111	1	All	E22
E1	Shunt limiter current	PWR	HLA	0 to 10 amps	95	1	All	45	All	All	90	1	All	E1
E2	Battery current	PWR	HLA	-10 to +5 amps	95	2	All	46	All	All	90	2	All	E2
E3	Main bus current	PWR	HLA	0 to 10 amps	95	3	All	3 +16n (n = 0–7)	All	All	90	3	All	E3
E4	Regulated bus current	PWR	HLA	0 to 5 amps	95	4	All	2-	All	All	93	1	All	E4
E5	Main bus voltage	PWR	HLA	0 to 40 volts	95	5	All	49	All	All	93	2	All	E5
E6	Regulated bus voltage	PWR	HLA	0 to 30 volts	95	6	All	50	All	All	93	3	All	E6
E15	Total solar current	PWR	HLA	0 to 10 amps	94	4	All	121	All	All	110	5	All	E15
E17	Solar paddle 1 position (SAGE)	PWR	HLA	-90 to +90°	94	5	All	N/A			96	6	All	E17
E18	Solar paddle 2 position (SAGE)	PWR	HLA	-90 to +90°	94	6	All	N/A			96	7	All	E18
F3	Pyro arm (A or B)	PWR	BL	1=Arm	104	8	2	77	8	2	109	8	2	F3
F4	Battery on/off (A and B)	PWR	BL	0=On	104	1	2	77	1	2	109	1	2	F4
F5	Battery thermostat enable/dis	PWR	BL	1=Enable	104	1	3	77	1	3	109	1	3	F5
F6	UV/OI sensor bypass (28-volt)	PWR	BL	1=Off	104	1	4	77	1	4	109	1	4	F6
F7	UV/OI flag set/reset	PWR	BL	1=Set	104	1	5	77	1	5	109	1	5	F7
F8	Solar string 1 on/off	PWR	BL	1=On	104	1	6	77	1	6	109	1	6	F8
F9	Solar string 2 on/off	PWR	BL	1=On	104	1	7	77	1	7	109	1	7	F9
F10	Solar string 3 on/off	PWR	BL	1=On	104	1	8	77	1	8	109	1	8	F10
F11	Voltage regulator bypass on/off	PWR	BL	1=On	104	2	6	77	2	6	109	2	6	F11
F12	Voltage limiter A on/off	PWR	BL	1=On	104	2	7	77	2	7	109	2	7	F12
F13	Voltage limiter B on/off	PWR	BL	1=On	104	2	8	77	2	8	109	2	8	F13
F14	Solar-array drive enable/dis	PWR	BL	1=Enable	104	3	5	N/A			109	3	5	F14

Table 3-1 (Continued)

ID	Measurement Name	Subsys.	Type	Parameter Range	Launch Word	Launch Frame	Launch Bits	HCMM Orbital Word	HCMM Orbital Frame	HCMM Orbital Bits	SAGE Orbital Word	SAGE Orbital Frame	SAGE Orbital Bit	ID
F41	Pressure XDR on/off	OAS	BL	1=On	104	6	1	77	6	1	N/A			F41
F42	Burn enable 1	OAS	BL	1=Enabled	104	7	2	77	7	2	N/A			F42
F43	Burn enable 2	OAS	BL	1=Enabled	104	7	1	77	7	1	N/A			F43
F44	Burn enable 3	OAS	BL	1=Enabled	104	6	8	77	6	8	N/A			F44
F45	Burn enable 4	OAS	BL	1=Enabled	104	6	7	77	6	7	N/A			F45
F46	Burn enable 5	OAS	BL	1=Enabled	104	6	6	77	6	6	N/A			F46
F47	Burn enable 6	OAS	BL	1=Enabled	104	6	5	77	6	5	N/A			F47
F48	Burn enable 7	OAS	BL	1=Enabled	104	6	4	77	6	4	N/A			F48
F49	Burn enable 8	OAS	BL	1=Enabled	104	6	3	77	6	3	N/A			F49
F50	Burn enable 9	OAS	BL	1=Enabled	104	6	2	77	6	2	N/A			F50
F51	OAS engine valve A on/off	OAS	BL	1=On	104	7	3	77	7	3	N/A			F51
F52	OAS engine enable/disable	OAS	BL	1=Enabled	104	7	4	77	7	4	N/A			F52
F53	Thruster heater 1 on/off	OAS	BL	1=On	104	7	5	77	7	5	N/A			F53
F54	Thruster heater 2 on/off	OAS	BL	1=On	104	7	6	77	7	6	N/A			F54
F55	Valve heater on/off	OAS	BL	1=On	104	7	7	77	7	7	N/A			F55
F56	Line heater on/off	OAS	BL	1=On	104	7	8	77	7	8	N/A			F56
F57	Tank heater on/off	OAS	BL	1=On	104	8	6	77	8	6	N/A			F57
F58	Tank thermostat bypass	OAS	BL	1=	104	8	7	77	8	7	N/A			F58
F59	Line thermostat bypass	OAS	BL	1=	104	8	8	77	8	8	N/A			F59
F60	OAS engine valve B on/off	OAS	BL	1=On	104	8	3	77	8	3	N/A			F60

Table 3-1 (Continued)

ID	Measurement Name	Subsys.	Type	Parameter Range	Launch Word	Launch Frame	Launch Bits	HCMM Orbital Word	HCMM Orbital Frame	HCMM Orbital Bits	SAGE Orbital Word	SAGE Orbital Frame	SAGE Orbital Bits	ID
T18	CMD demod. data present	CDH	BL	1=Present	47	All	1	47	All	1	47	All	1	T18
T28	Ranging on/off	CDH	BL	1=On	47	All	4	N/A				All	4	T28
T25	Phase lock status	CDH	BL	1=Locked	47	All	3	N/A				All	3	T25
T29	Coherent/noncoherent	CDH	BL	1=Coherent	47	All	2	N/A				All	2	T29
F15	Sequencer power on/off	CDH	BL	1=On	104	3	6	77	3	6	109	3	6	F15
F16	Sequencer hold on/off	CDH	BL	1=On	104	3	7	77	3	7	109	3	7	F16
F17	Sequencer off zero	CDH	BL	1=Off zero	104	3	8	77	3	8	109	3	8	F17
F18	S-band XMTR on/off	CDH	BL	1=On	104	4	1	74	All	1	109	4	1	F18
F19	PCM encoder operate/standby	CDH	BL	1=Operate	104	4	2	77	4	2	109	4	2	F19
F20	PCM bit rate 1024/8192	CDH	BL	1=1024	104	4	3	77	4	3	109	4	3	F20
F21	PCM format 1	CDH	BL	1=Format 1	104	4	4	77	4	4	109	4	4	F21
F22	PCM format (2 HCMM, 3 SAGE)	CDH	BL	1=Format 2/3	104	4	5	77	4	5	109	4	5	F22
F23	PCM format 4	CDH	BL	1=Format 4	104	4	6	77	4	6	109	4	6	F23
F24	64 Hz/2 Hz clock	CDH	BL	1=64 Hz	104	4	7	77	4	7	109	4	7	F24
F25	XPDR XMTR on/off	CDH	BL	1=On	104	3	4	N/A			109	3	4	F25
T7	Spacecraft clock	CDH	SER	1 sec to 194 days	14-16	All	All	14-16	All	All	14-16	All	All	T7
T16	Command clock	CDH	SER	1 sec to 24 hours	30-31	All	All	30-31	All	All	30-31	All	All	T16
T8	Command replica	CDH	SER	See 5.7	N/A			61-66	All	All	61-66	All	All	T8
T10	Command execution counter	CDH	SER	0 to 255	32	All	All	32	All	All	32	All	All	T10
T17	Command rejection counter	CDH	SER	0 to 255	24	All	All	29	All	All	29	All	All	T17
T1	Frame sync	CDH	INT	$171_8/104_8/334_8$	126-128	All	All	126-128	All	All	126-128	All	All	T1
T2	Frame ID (minor frame counter)	CDH	INT	1 to 8 (HCMM/Launch/SAGE	8	All	All	13	All	All	13	All	All	T2
M1	Tank pressure	OAS	HLA	0 to 500 PSIA	96	1	All	44	All	All	N/A			M1
M5	Tank temp. (ref. junct. temp.)	OAS	HLA	-30 to +100°C	110	4	All	93	5	All	N/A			M5
M4	Line temp.	OAS	HLA	-30 to +100°C	110	3	All	78	3	All	N/A			M4
M2	Valve temp.	OAS	HLA	-30 to +100°C	110	1	All	78	1	All	N/A			M2
M3	Thruster temp.	OAS	LLA	+10 to +1600°C	110	2	All	78	2	All	N/A			M3

SECTION 4

TELEMETRY FORMAT MATRICES

4.1 GENERAL

This section contains matrices of launch, HCMM, and SAGE formats (Figures 4-1 through 4-9), including subcommutation and bi-level assignments. These are intended to provide a visual picture of the PCM formats for user reference. The Memory Dump Format matrix is discussed in Section 8.

4.2 TIMING

Ground-time correlation of real-time data will generally be referenced to the end of the 24-bit frame synchronization pattern.

Internal to the encoder, the format functions are generated one bit period earlier than that which is transmitted. Major and minor frame synchronization pulses are generated for spacecraft equipment during internal word 124. This will be further discussed in appropriate following sections.

Ground-time correlation of SAGE tape recorder data will probably be referenced to the spacecraft clock (T7) as correlated to ground time from bounding, real-time passes. The spacecraft clock has a resolution of 1 second, is independent of bit rate, and operates continously.

FRAME ID

TLM MODE (T30 + T13)

CMD MODE (T15, ETC.)

FRAME SYNC

						1										
SP	E4	E3	A1	A2	A3	SP	D28	D28	D28	D28	D28	T2	T7	T7	T7	16
A5	A6	E3	A7	A8	D29		SP	SP	SP	SP	SP	T17	T16	T16	T10	32
A9	A10	E3	A1	A2	A3		A11	A12	A14	A15	M1	E1	E2	BL16	BL9	48
E5	E6	E3	SP	T3	T4		T5	SP	BL10	BL11	BL12	T8	T8	T8	T8	64
T8	T8	E3	A1	A2	A3		BL13	BL14	BL15	SP	SP	SC1	SC2	SC3	SC4	80
D1	D2	E3	D3	D4	D5		D6	D7	D8	D9	D10	SC5	SC6	SP	SP	96
D11	D12	E3	A1	A2	A3		D13	D14	D15	D16	D17	SP	SP	A27	A21	112
D18	A20	E3	A22	A23	A24	SP	A28	E15	A29	A30	A31	A41	T1	T1	T1	128

1 Words 7, 23, 39, 55, 71, 87, 103, and 119 are supercommutated to one spare HLA analog.

Figure 4-4. Minor Frame, HCMM Orbital Mode Format (PCM Format 2)

	(MSB) BIT1	BIT2	BIT3	BIT4	BIT5	BIT6	BIT7	(LSB) BIT8	WORD/FRAME
BL1	F32	F4	F5	F6	F7	F8	F9	F10	77/1
BL2	F28	F29	F27	F30	F31	F11	F12	F13	77/2
BL3	SP	F62	F61	SP	SP	F15	F16	F17	77/3
BL4	SP	F19	F20	F21	F22	F23	F24	F26	77/4
BL5	F40	F39	F38	F37	F36	F35	F34	F33	77/5
BL6	F41	F50	F49	F48	F47	F46	F45	F44	77/6
BL7	F43	F42	F51	F52	F53	F54	F55	F56	77/7
BL8	SP	F3	F60	F1	F2	F57	F58	F59	77/8
BL9	T30 B-2	T30 B-1	T30 B-0	T13 B-4	T13 B-3	T13 B-2	T13 B-1	T13 B-0	48/All
BL10	A35	A34	A32	A42	A19	A18 B-1	A18 B-2	A18 B-3	58/All
BL11	A16 B-8	A16 B-7	A16 B-6	A16 B-5	A16 B-4	A16 B-3	A16 B-2	A16 B-1	59/All
BL12	A17 B-8	A17 B-7	A17 B-6	A17 B-5	A17 B-4	A17 B-3	A17 B-2	A17 B-1	60/All
BL13	D43	D42	D41	D40	D39	D38	D37	D36	72/All
BL14	SP	SP	D49	D48	D47	D46	D45	D44	73/All
BL15	F18	SP	SP	SP	SP	SP	S17	S18	74/All
BL16	T18	SP	SP	SP	T15	T15	T15	T15	47/All

Figure 4-6. Bilevel Channels, HCMM Orbital Mode Format (PCM Format 2)

SUBCOM →

MINOR FRAME ↓	SC 1	SC 2	SC 3	SC 4	SC 5	SC 6
1	BL1	M2	T11	D19	D27	E22
2	BL2	M3	T12	D20	T14	SP
3	BL3	M4	S1	D21	A25	SP
4	BL4	E7	S2	D22	A26	SP
5	BL5	E8	S3	D23	M5	SP
6	BL6	E9	S4	D24	T20	SP
7	BL7	E10	S5	D25	T21	SP
8	BL8	E11	S6	D26	SP	SP
MAIN COM WORD	77	78	79	80	93	94

Figure 4-5. Subcommutated Channels, HCMM Orbital Mode Format (PCM Format 2)

SECTION 5

BASE-MODULE PCM DIGITAL DATA DESCRIPTION

5.1 GENERAL

This section provides information for reducing of Base-Module serial and internal multibit digital data. Items are identified by their format ID (Sections 3 and 4) and are presented in sequences of association.

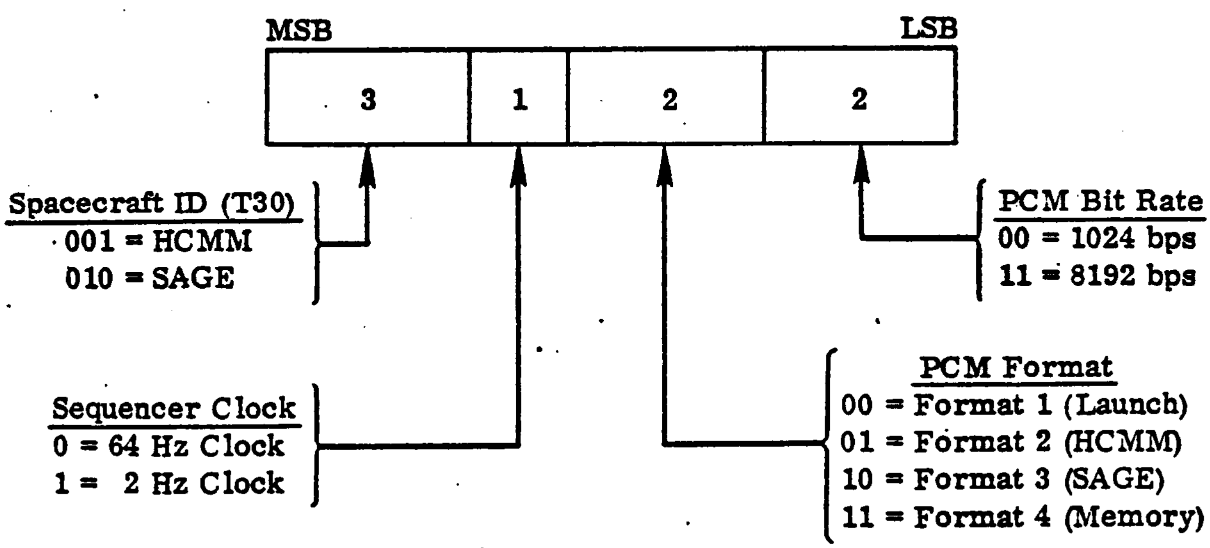
Except when required for validation or qualification, single bilevel flags are not included. (See Section 3 for equivalences of "1" and "0" states.) Instrument digital data are discussed in Section 6 (HCMM) and Section 7 (SAGE).

Sun-sensor identification and angle conversion tables are given in Appendix A.

5.2 FRAME SYNCHRONIZATION (T1)

MSB		LSB
8 BITS	8 BITS	8 BITS
171_8	104_8	334_8
01111001	01000100	11011100

5.3 TELEMETRY MODE (T13) AND SPACECRAFT ID (T30)



Flag measurements F20 through F24 indicate the states of the relays in the base-module relay unit, that control the encoder modes. However, for normal data reduction, use the above-described T13 measurement.

Table 2-1 lists valid bit rates for each format.

5.4 SPACECRAFT CLOCK (T7)

MSB LSB

8	8	8

This clock represents a full binary count to 194 days in 1-second increments independent of bit rate. The clock operates continuously.

5.5 FRAME ID (MINOR-FRAME COUNTER) (T2)

MSB LSB

3	5

Don't care

XX000 → XX111 Formats 1, 2, 3
00000 → 11111 Format 4

T2 is the minor-frame counter, or identifier. The count is one less than the frame numeralization indicated in Sections 3 and 4. For example, T2 will read "XX100" for minor frame 5 in formats 1, 2, and 3.

Sections 7 and 10 describe the identification for subcommutation internal to the SAGE instrument and tape recorders.

SECTION 9

HCMM PCM ANALOG DATA CONVERSION FACTORS

9.1 GENERAL

This section is concerned with the conversion of HCMM analog data that is telemetered by way of the Base Module PCM encoder. Conversions are provided for both HCMM Base Module and HCMM Instrument Module analog performance parameters.

> NOTE
>
> Because analog performance parameter calibrations will change slightly from mission to mission, SAGE Base Module analog parameter conversions are provided in Section 10.

9.2 SCALE FACTOR

The Base Module PCM encoder converts all analog input channels to 8-bit digital equivalents with a nominal resolution of 0.02 volts/digital increment. Three encoder calibration points are provided in the PCM data as identified in Table 9-1.

Table 9-1
Encoder Internal Calibration

ID	Nominal Voltage	Nominal Readout (decimal)
T3	0.11	5
T4	2.51	125
T5	5.01	250

The accuracy, linearity, and repeatibility of the calibration levels indicated in Table 9-1 are to be verified in test. However, all users of HCMM data should be prepared to accommodate deviations from the norm. Such deviations include scale changes, nonlinearity, and zero-reference offset.

Users may use any option they choose to properly scale the telemetered data. However, for the purposes of this document and to provide common ground for all users in implementing the conversion factors in Table 9-2, the following definitions apply:

- D = Decimal equivalent of the raw binary data
- S = Scale factor derived from encoder internal calibrations
- Z = Zero offset in volts
- V = (D x S) - Z = scaled raw binary data in volts

For example:

Assumptions:

- Raw data (D) = 11000011
- T3 = 00000100
- T5 = 11111011
- A/D conversion is linear over full range

The decimal equivalents are:

- Raw data = 195
- T3 = 4
- T5 = 251

The scaling slope (S) is determined by:

$$S = \frac{\text{T5 (nominal volts)} - \text{T3 (nominal volts)}}{\text{T5 (decimal)} - \text{T3 (decimal)}} =$$

$$\frac{5.01 - 0.11}{251 - 4} = \frac{4.90}{247} = 0.0198 \text{ volts/decimal increment}$$

It can also be seen that the actual A/D zero (Z) is offset to -0.0198 volts. Therefore, the scaling of raw data, D, would be as follows:

$$(195 \times 0.0198) - (-0.0198) = 3.881 \text{ volts}$$

.3 HCMM ANALOG CONVERSION TABLE

The analog performance parameter conversions for HCMM are listed in Table 9-2. Explanations of the column headings in the table are as follows:

- ID : The parameter ID in Table 3-1 and 3-2
- Measurement Name: Self-explanatory
- Scale factor : Algorithm to be used in conjunction with "V" (see Section 9.2)
- Units : Engineering units or other nomenclature
- Remarks : Validation criteria (such as F36 = "1"), etc.

Scale factor constants are carried out to a suggested number of decimal places to accommodate changes during the program. Users may implement whatever decimal ranges meet their requirements.

Temperature sensor curves are contained in Appendix B and are referenced in Table 9-2 by curve type, such as Type A.

Table 9-2
HCMM Analog Conversions

ID	Measurement Name	Scale Factor	Units	Remarks
A1	Magnetometer X (roll)	V (0.240) - 0.600	Gauss	
A2	Magnetometer Y (pitch)	V (0.240) - 0.600	Gauss	Valid only when A13 & A32 & A34 = 1
A3	Magnetometer Z (yaw)	V (0.240) - 0.600	Gauss	
A5	Electromagnet bias X (roll)	V (4,000) - 10,000	Pole-cm	
A6	Electromagnet bias Y (pitch)	V (4,000) - 10,000	Pole-cm	
A7	Electromagnet bias Z (yaw)	V (4,000) - 10,000	Pole-cm	
A8	Electromagnet moment (roll)	V (4,000) - 10,000	Pole-cm	
A9	Electromagnet moment (pitch)	V (4,000) - 10,000	Pole-cm	
A10	Electromagnet moment (yaw)	V (4,000) - 10,000	Pole-cm	
A27	Roll error (fine)	V (2.00) - 5.00	Deg	
A11	Roll error (course)	V (20.0) - 50.0	Deg	
A28	Pitch error (fine)	V (0.80) - 2.00	Deg	
A12	Pitch error (course)	V (20.0) - 50.0	Deg	Valid only when A19 = 1
A14	Scan wheel B speed	V (440) - 0.0	Rpm	
A13	Scan wheel A speed	V (440) - 0.0	Rpm	
• A15	Roll bias	V (2.00) - 5.00	Deg	
A24	SWA-B H1 Duty cycle	V (10.0) - 0.0	%	
A29	SWA-B H2 Duty cycle	V (10.0) - 0.0	%	
A25	Scan wheel B temp.	Type C curve	°C	
A30	SWA-B Motor excitation	V (6.60) - 0.0	Volts	
A31	Power supply temp.	Type C curve	°C	
A26	5-volt supply	V (1.00) - 0.0	Volts	
A41	10-volt supply	V (1.00) - 0.0	Volts	
A21	Magnetic field rate (roll)	V (0.0048) - 0.120	Gauss/sec	
A22	Magnetic field rate (pitch)	V (0.0048) - 0.120	Gauss/sec	Valid only when A18 & A32 & A34 = 1
A23	Magnetic field rate (yaw)	V (0.0048) - 0.120	Gauss/sec	
A20	Spin accelerometer	V (44.15) - 8.83	Rpm	Requires F2 = 1
S7	Stage 4 pressure	V (187.5) - 0.0	PSI	
S8	Stage 4 Z acceleration	V (6.20) - 1.00	Gs	
S9	Stage 4 X vibration	V (0.200) - 0.500	Gs	Valid only when F28 = 1
S10	Stage 4 Y vibration	V (0.200) - 0.500	Gs	
S11	Stage 4 Z vibration	V (0.200) - 0.500	Gs	
E10	+X solar-paddle temp. 1	V (-39.4) - 97.0	°C	
E11	+X solar-paddle temp. 2	Type A curve	°C	
E8	-X solar-paddle temp. 1	Type A curve	°C	
E9	-X solar-paddle temp. 2	V (-39.4) - 97.0	°C	
E7	Battery temp. 1	Type A curve	°C	
E22	Battery temp. 2	Type A curve	°C	
E1	Shunt limiter current	V (2.00) - 0.00	Amps	
E2	Battery current	V (4.00) - 10.00	Amps	

Table 9-2 (Continued)

ID	Measurement Name	Scale Factor	Units	Remarks
E3	Main-bus current	V (2.00) - 0.00	Amps	
E4	Regulated-bus current	V (1.00) - 0.00	Amps	
E5	Main-bus voltage	V (3.00) - 0.00	Volts	
E6	Regulated-bus voltage	V (6.00) - 0.00	Volts	
E13	Total solar current	V (2.00) - 0.00	Amps	Includes external power
S1	Equipment panel 1 temp.	Type A curve	°C	
S2	Equipment panel 2 temp.	Type A curve	°C	
S3	Equipment panel 3 temp.	Type A curve	°C	
S4	Equipment panel 4 temp.	Type A curve	°C	
S5	Equipment panel 5 temp.	Type A curve	°C	
S6	Scan wheel B mount temp.	Type A curve	°C	
T11	S-band XMTR temp.	Type A curve	°C	
T20	S-band XMTR RF PWR	V (0.40) - 0.00	Watts	Valid only when F13 = 1
T12	VHF XMTR temp.	Type A curve	°C	
T21	VHF XMTR PWR	V (0.050) - 0.00	Watts	Valid only when F = 1
T14	VHF RCVR AGC	V (0.80) - 0.00	Volts	
M1	Tank pressure	TBD	PSI	Valid only when F41 = 1
M5	Tank temp. (ref. junc. temp.)	TBD	°C	
M4	Line temp.	TBD	°C	
M2	Valve temp.	TBD	°C	
M3	Thruster temp.	TBD	°C	
D1	Blackbody temp. 1	56.6 - V (10.31)	°C	
D2	Blackbody temp. 2	56.6 - V (10.31)	°C	
D3	IR preamp pwr	V (1.00) - 0.0	Volts	
D4	Offset voltage	V (2.0) - 14.33	Volts	
D5	Elex. current	V (0.238) - 0.0	Amps	
D6	Motor drive current	V (0.270) - 0.0	Amps	
D7	Scan motor speed	360 - V (35.6)	Rpm	
D8	Compensation motor speed	5803 - V (744.6)	Rpm	Valid only when F33 = 1
D9	Patch power	V^2 (0.33) - 0.0	Milliwatts	
D10	Telemetry power	V (4.425) - 0.0	Volts	
D11	Cone cover position	V (22.73) - 4.92	Degrees	
D12	Purge pressure	150 - V (30.48)	PSIG	
D13	+15-volt monitor	V (4.425) - 0.0	Volts	
D14	-15-volt monitor	V (4.006) - 23.39	Volts	
D15	+5-volt monitor	V (2.0) - 0.0	Volts	
D16	Patch temp.	V (6.21) - 105.68	K	
D17	Cooler wall HSG temp.	Type D Curve	K	

Table 9-2. (Continued)

ID	Measurement Name	Scale Factor	Units	Remarks
D18	N/A			
D19	Elex. temp.	56.6 - V (10.31)	°C	Valid only when F33 = 1
D20	Baseplate temp.	56.6 - V (10.31)	°C	
D21	Cone temp.	143.65 - V (30.59)	°K	
D22	Motor HSG temp.	Type E curve	°C	
D23	VCO temp.	Type F curve	°C	Valid only when F34 = 1
D24	HCMR +Z temp. 1	Type F curve	°C	
D25	HCMR +Z temp. 2	Type F curve	°C	
D26	Instr. mod. structure temp.	Type F curve	°C	
D27	N/A			

APPENDIX A

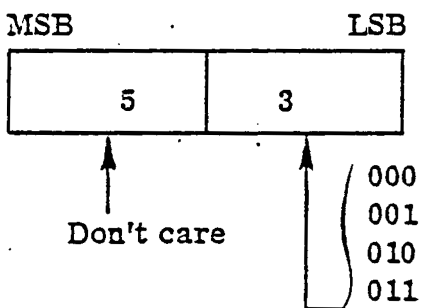
SUN-SENSOR ANGLE CONVERSION TABLES

A-1 INTRODUCTION

This appendix describes conversions for AEM Base Module Sun-sensor digital data. These data are located in three 8-bit telemetry words:

- A18 : Sun-sensor ID
- A16 : Sun-sensor reticle A
- A17 : Sun-sensor reticle B

A-2 SUN-SENSOR ID (A18)



	HCMM	SAGE
000 =	N/A	N/A
001 =	#1	#1
010 =	#2	#2
011 =	#3	#3
100 =	N/A	#4
101 =	N/A	#5
110 =	N/A	N/A
111 =	No Sun	No Sun

A18 is valid only when F1 = 1.

A-3 SUN-SENSOR RETICLES (A16 and A17)

Each Sun sensor has two reticles which provide 8-bit gray-coded angle data. Reticle A data appears in A16 and Reticle B data appears in A17. Preliminary conversion of these data is the subject of this Appendix and is independent of sensor mounting and alignment. Conversion of these data to spacecraft references is not a subject of this appendix.

A gray code to Base 10 conversion table is provided in Table A-1. This lookup table may be used as in the following example:

Convert Gray Code A16 = 235_8 to Decimal

1. Go down left-hand octal column to "230"
2. Go across to octal column "XX5"
3. Decimal equivalent = 233

Another conversion method may be used to avoid maintaining a large lookup table in the operating system. This method is as follows:

- Starting with the MSB (2^7), determine = 0 or 1
- If 0, 2^7 bit = 0, go to next bit (2^6)
  If 1, 2^7 bit = 1, invert next bit (2^6)
- If 2^6 bit now = 0, 2^6 bit remains 0, go to next bit (2^5)
  If 2^6 bit now = 1, 2^6 bit remains 1, invert next bit (2^5)
- Etc., etc.

Having completed conversion of reticle data to decimal values, conversion to sensor angles may then be accomplished. The formula for this conversion is:

0.5 (Decimal) - 63.75 Degrees

A16 and A17 are valid only when F1 = 1.

Table A-1
Octal Gray Code to Base 10 Conversion

Oct	Dec	Oct	Dec	Oct	Dec	Oct	Dec	Oct	Dec	Oct	Dec	Oct	Dec	Oct	Dec
000	0000.	XX1	0001.	XX2	0003.	XX3	0002.	XX4	0007.	XX5	0006.	XX6	0004.	XX7	0005.
010	0015.	XX1	0014.	XX2	0012.	XX3	0013.	XX4	0008.	XX5	0009.	XX6	0011.	XX7	0010.
020	0031.	XX1	0030.	XX2	0028.	XX3	0029.	XX4	0024.	XX5	0025.	XX6	0027.	XX7	0026.
030	0016.	XX1	0017.	XX2	0019.	XX3	0018.	XX4	0023.	XX5	0022.	XX6	0020.	XX7	0021.
040	0063.	XX1	0062.	XX2	0060.	XX3	0061.	XX4	0056.	XX5	0057.	XX6	0059.	XX7	0058.
050	0053.	XX1	0049.	XX2	0051.	XX3	0050.	XX4	0055.	XX5	0054.	XX6	0052.	XX7	0053.
060	0042.	XX1	0033.	XX2	0035.	XX3	0034.	XX4	0039.	XX5	0038.	XX6	0036.	XX7	0037.
070	0047.	XX1	0046.	XX2	0044.	XX3	0045.	XX4	0040.	XX5	0041.	XX6	0043.	XX7	0042.
100	0127.	XX1	0126.	XX2	0124.	XX3	0125.	XX4	0120.	XX5	0121.	XX6	0123.	XX7	0122.
110	0112.	XX1	0113.	XX2	0115.	XX3	0114.	XX4	0119.	XX5	0118.	XX6	0116.	XX7	0117.
120	0096.	XX1	0097.	XX2	0099.	XX3	0098.	XX4	0103.	XX5	0102.	XX6	0100.	XX7	0101.
130	0124.	XX1	0110.	XX2	0108.	XX3	0109.	XX4	0104.	XX5	0105.	XX6	0107.	XX7	0106.
140	0064.	XX1	0065.	XX2	0067.	XX3	0066.	XX4	0071.	XX5	0070.	XX6	0068.	XX7	0069.
150	0079.	XX1	0078.	XX2	0076.	XX3	0077.	XX4	0072.	XX5	0073.	XX6	0075.	XX7	0074.
160	0095.	XX1	0094.	XX2	0092.	XX3	0093.	XX4	0088.	XX5	0089.	XX6	0091.	XX7	0090.
170	0080.	XX1	0081.	XX2	0083.	XX3	0082.	XX4	0087.	XX5	0086.	XX6	0084.	XX7	0085.
200	0255.	XX1	0254.	XX2	0252.	XX3	0253.	XX4	0248.	XX5	0249.	XX6	0251.	XX7	0214.
210	0240.	XX1	0241.	XX2	0243.	XX3	0242.	XX4	0247.	XX5	0246.	XX6	0244.	XX7	0245.
220	0224.	XX1	0225.	XX2	0227.	XX3	0226.	XX4	0231.	XX5	0230.	XX6	0228.	XX7	0229.
230	0239.	XX1	0238.	XX2	0236.	XX3	0237.	XX4	0232.	XX5	0233.	XX6	0235.	XX7	0234.
240	0192.	XX1	0193.	XX2	0195.	XX3	0194.	XX4	0199.	XX5	0198.	XX6	0196.	XX7	0197.
250	0207.	XX1	0206.	XX2	0204.	XX3	0205.	XX4	0200.	XX5	0201.	XX6	0203.	XX7	0202.
260	0223.	XX1	0222.	XX2	0220.	XX3	0221.	XX4	0216.	XX5	0217.	XX6	0219.	XX7	0218.
270	0208.	XX1	0209.	XX2	0211.	XX3	0210.	XX4	0215.	XX5	0214.	XX6	0212.	XX7	0213.
300	0128.	XX1	0129.	XX2	0131.	XX3	0130.	XX4	0135.	XX5	0134.	XX6	0132.	XX7	0133.
310	0143.	XX1	0142.	XX2	0140.	XX3	0141.	XX4	0136.	XX5	0137.	XX6	0139.	XX7	0138.
320	0159.	XX1	0158.	XX2	0156.	XX3	0157.	XX4	0152.	XX5	0153.	XX6	0155.	XX7	0154.
330	0144.	XX1	0145.	XX2	0147.	XX3	0146.	XX4	0151.	XX5	0150.	XX6	0148.	XX7	0149.
340	0191.	XX1	0190.	XX2	0188.	XX3	0189.	XX4	0184.	XX5	0185.	XX6	0187.	XX7	0186.
350	0176.	XX1	0177.	XX2	0179.	XX3	0178.	XX4	0183.	XX5	0182.	XX6	0180.	XX7	0181.
360	0160.	XX1	0161.	XX2	0163.	XX3	0162.	XX4	0167.	XX5	0166.	XX6	0164.	XX7	0165.
370	0175.	XX1	0174.	XX2	0172.	XX3	0173.	XX4	0168.	XX5	0169.	XX6	0171.	XX7	0170.

APPENDIX D - SPECIALIZED TELEMETRY PROCESSOR FUNCTIONS

This appendix describes four specialized capabilities of the telemetry processor functions. Section D. 1. describes recovery from NAMELIST errors in interactive mode; Section D.2. describes extensive processing control display; Section D.3. describes error message display, and Section D.4. describes execution time determination of telemetry data processing. The MSAD/AEM-A development task will decide whether or not the first three options are implemented.

D.1 RECOVERY FROM NAMELIST ERRORS IN INTERACTIVE MODE

When FORTRAN IBCOM error handling routines detect syntax errors (incorrectly spelled variable names, misplaced commas or blanks, variable names not defined in the appropriate NAMELIST, etc.) while reading NAMELIST data sets, a message including the name of the variable detected is printed, but the remainder of the input variables in the NAMELIST data set are not printed, and processing continues with no indication to the user of the detected error. The NAMELIST variables following the one in error are not read. Consequently, the graphics user has no indication, when the NAMELIST displays appear, that a syntax error has been made. To circumvent this problem, the capability is provided to intercept any error detected by IBCOM and to inform the user graphically that a NAMELIST error has been detected and on which NAMELIST display the incorrect parameter or syntax error is displayed. With this capability, the user will know that (1) all parameter variables or NAMELIST displays occurring prior to the one indicated in the error message have been initiated properly, (2) some parameters on the NAMELIST display indicated have not been initialized, and (3) none of the parameters on succeeding NAMELIST displays have been initialized and hence must be initialized interactively. The following paragraphs illustrate what occurs when the error procedure is invoked.

An example of a NAMELIST definition is given below where, for convenience of illustration, the names of all parameters on the first display (called NAME1) end in 1, all on the second display (NAME2) with 2, and all on the third (NAME3) with 3. Note that the order of occurrence of the parameter in the NAMELIST definition is irrelevant, but in the NAMELIST data set the items must occur in the order in which they are displayed in the GESS NAMELIST displays.

```
LOGICAL *1 $ ERROR (8)
NAMELIST/TEST NAME/$ ERROR, ABC1, DEF3, GHI2, JKL2, MNO1,
                                   PQR3, STU1, VWX3
```

The NAMELIST data set might then appear as follows:

```
Col. 2
 ↓
&TESTNAME
$ERROR = 'NAME1ƀƀƀ',
ABC1 = 1.3, MNO1 = 93, STU1 = 203.4,
$ERROR = 'NAME2ƀƀƀ',
GHI2 = 2.0, JKL2 = 3,,
$ERROR = 'NAME3ƀƀƀ',
DEF3 = 2.4, PQQ3 = 4.7, VWX3 = 7.6,
&END
```

Note that there are two syntax errors in this NAMELIST data set: there are two commas after JKL2, and PQR3 is misspelled as PQQ3. FORTRAN IBCOM will first encounter the two consecutive commas and terminate reading the data set. With the NAMELIST error recovery capability, a facsimile of the following error message is displayed:

AN ERROR HAS BEEN DETECTED READING THE TESTNAME NAMELIST. THE ERROR OCCURRED INITIALIZING DISPLAY NAME2ƀƀƀ. PRESS ALT-END TO CONTINUE.

The user is hence informed that (1) an error has occurred, and (2) that all parameters on display NAME1 have been initialized, that at least one parameter on display NAME2 has not been initialized, and that none are initialized on display NAME3. The user can then examine and initialize them interactively as desired.

In noninteractive mode, the same message is produced in hardcopy and the job is terminated.

D.2 EXTENSIVE PROCESSING CONTROL DISPLAY

In interactive mode, the processing loop control display capability allows generation of a display in GESS GO status during all extensive central processing unit (CPU) and/or I/O processing. Its purpose is to inform the user continually of the status of processing and to permit termination of a processing loop that exceeds prescribed limits. The second capability is also used in noninteractive mode. The display is shown in Figure D-1.

In interactive mode, the display appears each time a large processing loop is entered. The first parameter on the display tells which loop is being executed (each loop is assigned a unique number). The display is then updated at 30-second intervals. The second and fourth parameters displayed provide the limiting parameters for the loop: the amount of CPU time allowed for the loop, and the number of times the loop is allowed to be executed. The third and fifth parameters displayed provide the current value of the elapsed CPU time and the number of times the loop has been executed.

Each time the loop is executed (generally much more often than every 30 seconds), a dummy GESS CPOINT is automatically invoked to allow intervention via program function key 30, causing the GESS XSTOPS display to appear. The LOOPS display is then placed in STOP status so that a SKIP operations causes

```
LOOPS

                 LARGE LOOP CONTROL DISPLAY

THE SYSTEM IS IN LOOP NUMBER . . . . . . . . . . . . .   3

THIS DISPLAY WILL BE UPDATED AT
30-SECOND INTERVALS.

NUMBER OF CPU MINUTES ALLOWED FOR LOOP . . . . . . . .   0.1E 02

NUMBER OF ELAPSED CPU MINUTES IN LOOP  . . . . . . . .   0.3E-83

NUMBER OF ALLOWED LOOPS  . . . . . . . . . . . . . . .   10

NUMBER OF ELAPSED LOOPS  . . . . . . . . . . . . . . .   2

PRESS KEY 30 TO INTERRUPT LOOP;
SET STATUS OF THIS DISPLAY TO 'STOP';
THEN SKIP FROM XSTOPS.

DISCONTINUE LOOP (NO, YES) . . . . . . . . . . . . . .   NO
```

Figure D-1. Processing Loop Control Display

one more iteration and brings up the LOOPS display. The final parameter on the display can be changed from NO to YES, and a SKIP operation from the display then causes YES to change back to NO and STOP status to revert to GO. The current loop is exited to a user-defined point in the code.

When either limit is reached (allotted CPU time or number of allowed loops), the display status is automatically changed to STOP, and DISCONTINUE LOOP is changed to YES. If the user wishes to discontinue the loop, he skips from the display and program control exits the loop to a user-defined point in the code. (The status is automatically changed back to GO, and DISCONTINUE LOOP is changed to NO in preparation for the next loop encountered.) If the user chooses to continue the loop, he changes DISCONTINUE LOOP from YES back to NO and performs a SKIP operation. The display automatically returns to GO status and continues looping. The loop limits (second and fourth parameters displayed) are automatically changed to 9.0E10 seconds and 999999 loops, respectively.

In noninteractive mode, the loop is exited when either of the loop limits is reached, and hardcopy output on FORTRAN unit 6 denotes this occurrence.

In summary, the LOOPS display informs the user of the following:

- Whether he is locked out of the CPU by other jobs
- Whether the computer is down or in STOP status
- How much CPU time the loop has used
- How many times the loop has been executed
- Which loopis currently being executed
- How many loops are allowed and how much CPU time is allowed for the loop

and allows the user to

- Terminate the loop before the loop limits are reached
- Terminate the loop when the loop limits are reached interactively or noninteractively
- Continue execution, if the loop is abnormally exited, at a different part of executable code than if normally exited

The code required for invoking the LOOP display in a FORTRAN subroutine requires two statements, one immediately preceding the initial statement in the loop, and one following it. Two examples follow, one using a DO loop, and the other

- Example Using DO Loop Statement

```
          .
          .
          .
      CALL LOOPS1 (3, 10, 100.0)
      DO 10 I = 1, 100
          CALL LOOPS2 (99)
          .
          .
          .
10        CONTINUE
      normal continuation
          .
          .
          .
99    abnormal continuation
          .
          .
          .
```

- Example Using Branch Statements

```
        .
        .
        .
      I = 1
      CALL LOOPS1 (3, 10, 100.0)
5     CONTINUE
      CALL LOOPS2 (99)
        .
        .
        .
      I = I + 1
      IF (I.LE.100) GO TO 5
      Normal continuation
        .
        .
        .
99    Abnormal continuation
```

D.3 ERROR MESSAGE DISPLAY

The error message display contains the following information:

1. User error number
2. Name of the subroutine detecting the error
3. Probable cause of the error
4. Suggested corrective action
5. Probable result if no corrective action is taken

An example of an error message is shown in Figure D-2. This message would be produced by the following statement:

```
CALL ERRMSG ('0011, 'TELMDRVbb', 'NO SUN DATA#',
'AVAILABLE#', 'PRESS KEY 3 AND#', 'SET SUN1 TO .TRUE.#',
'NO SUN DATA AVAILABLE FOR ATTITUDE#')
```

All error messages are documented in numerical order in Reference 13, and are explained in adequate detail that the user, with minimal time and effort, can determine the severity of the error and other related details.

ERROR MESSAGE NUMBER	001
DETECTED BY SUBROUTINE	TLMDRV
PROBABLE CAUSE OF ERROR	NO SUN DATA AVAILABLE
SUGGESTED CORRECTIVE ACTION	PRESS KEY 3 AND SET SUN1 TO TRUE
RESULT OF ALT-END	NO SUN DATA FOR ATTITUDE

Figure D-2. Error Display Message

D.4 EXECUTION TIME DETERMINATION OF TELEMETRY DATA PROCESSING

Unpacking and conversion of telemetry data from both OCC and IPD records are performed in a manner determined by parameters included in a separate telemetry processing function NAMELIST data set. These parameters are initialized to default values in a BLOCK DATA program, and the NAMELIST containing them will nominally not be required. In the event of unexpected systematic anomalies in the telemetry data, however, this NAMELIST provides a means of modifying the unpacking and/or conversion process without creating a new load module. These modifications can be made, in fact, without terminating the current job. Examples of uses of this feature include unanticipated occurrences of

- Magnetometer mounted upside-down
- Magnetometer wired backwards
- A sign bit of 1 signifying positive rather than negative
- A negative sign bit implying the remaining bits be inverted
- Parameters occurring in unexpected positions in the telemetry record
- Parameters telemetered LSB first

In addition to the execution time modification of the unpacking and conversion, this feature allows the telemetry processing functions to be applied to other spacecraft with little or no modification, by supplying an appropriate NAMELIST data set.

The capabilities which are available for modification through NAMELIST minimally provide an ability to

- Determine the position within the telemetry record of all data items unpacked

- Invert all bits in a given byte
- Invert a given bit in a given byte
- Reverse the order of all bits in a given byte
- Reverse the order of an arbitrarily long string of bytes
- Pack from 1 to 4 arbitrary bytes into a full word, left padded with zeros, if necessary
- Invert a given bit of a given byte, based on a half-word mask
- Perform AND, INCLUSIVE OR, or EXCLUSIVE OR of a given byte with a mask byte
- Perform a test on the MSB of a given byte against a half-word comparator (0 or 1) and invert the remaining 7 bits if the test is met, as well as all 8 bits in an arbitrary number of additional arbitrary bytes

APPENDIX E - REVISED CALIBRATION DATA SUMMARY

This appendix contains an update to the calibration data in Appendix C and is the best information available as of January 26, 1978.

UNITED STATES GOVERNMENT

Memorandum

733:57:NJW:ch

TO : Richard S. Nankervis
Mission Support & Analysis Branch

DATE: August 8, 1977

FROM : Nicholas J. Witek
Electronics Systems Branch

SUBJECT: Response to Questions Concerning the Attitude Control System on the HCMM Spacecraft

REFERENCE: Memorandum to Mr. Britner from Mr. Nankervis, dated 6/22/77; Subject: AEM-A (HCMM) Telemetry Data

There exists in the HCMM Telemetry stream parameters which identify the fine and coarse scanwheel speed. They are as follows:

A14 - Scanwheel B Speed (fine)
[1710-2050 RPM]

A43 - Scanwheel B Speed
[340-2200 RPM]

These values are located in each minor frame of the HCMM Orbital Mode (Format 2). A43 is 8-bit byte #1, and A14 is 8-bit byte #42. The conversion equations for these parameters are shown in the attached Calibration Data Summary.

Responses to the questions posed by Dr. S.G. Hotovy of Computer Sciences Corporation are as follows:

1a. The input voltage V_0 depends only on the value of the quantity being measured.

1b. The 10-volt reference power supply is expected to vary from 9V to 11V.

2a. The input voltage V_0 depends only on the value of the quantity being measured .

2b. The 5-volt reference power supply is expected to vary from 4.5V to 5.0V.

2c. Since there are separate 5 and 10 volt power supplies, it is possible that they would vary independently. However the main source of power for these converters and regulators comes from the 28V bus of the spacecraft. Variations in the 28V bus might be reflected simultaneously in both the 5 and 10 volt supplies.

SUBJECT: Response to Questions Concerning the Attitude Control System on the HCMM Spacecraft

3. The signal to analog relationship is simply;

$$V = A\ V_0$$

4. The signal to analog conversion equations for the fine and coarse scanwheel speed, roll bias, scanwheel duty cycle II are shown in the attached Calibration Data Summary.

5a. The time-dependent conversion calibration levels (called out by T3. T4, T5) compensate for variations in the A/D converters in the PCM encoder.

5b. Determination of the absolute value of V_0 must take into account the variations in the 5 and 10 volt supplies telemetered by A26 and A41, and the calibration factors reflected in T3, T4, and T5.

5c. In addition to the conversion equation for a particular parameter, the overall calibration effect of T3, T4, and T5 must be incorporated to yield the absolute value of that parameter.

6. The position of the magnetometer has changed since the CDR. The new position on the S-Band antenna boom is shown in the attached mechanical drawing of the Spacecraft Orbital Configuration.

7. The magnetometer data that is put into the telemetry stream reflects the correction imposed by the Control Electronics Assembly.

Nicholas J. Witek

Nicholas J. Witek

Concurrence:

R O Britner

Ronald O. Britner
AEM PROJECT

Attachments (3)

cc: Mr. Britner, Code 406, w/attachs.
Mr. Grant, Code 733, w/attachs.
Mr. Williamson, Code 734, w/attachs.

TABLE 1.0-1: CALIBRATION DATA SUMMARY

NO.	DESIG-NATOR	MEASUREMENT NAME	CALIBRATION DATA REF.	FORMULA	REMARKS
1	A1	Magnetometer Roll	2.1	$g=(0.248V_{10}-V)/4.074$	V_{10} is Meas. A41 g in gauss
2	A2	Magnetometer Pitch	2.1	$g=(0.249V_{10}-V)/4.030$	g in gauss
3	A3	Magnetometer Yaw	2.1	$g=(0.249V_{10}-V)/4.033$	g in gauss
4	A5	E' magnet Bias X Roll	2.9	$V_{bx}=2V-5.00$	V_b in volts
5	A6	E' magnet Bias Y Pitch	2.9	$V_{by}=2V-5.00$	V_b in volts
6	A7	E' magnet Bias Z Yaw	2.9	$V_{bz}=2V-5.00$	V_b in volts
7	A8	E' magnet Moment Roll	2.1	$P_x=5000(V-0.242V_{10})$	P in pole-cm
8	A9	E' magnet Moment Pitch	2.1	$P_y=5000(V-0.242V_{10})$	P in pole-cm
9	A10	E' magnet Moment Yaw	2.1	$P_z=5000(V-0.240V_{10})$	P in pole-cm
10	A27	Roll Error, Fine	2.1.1	$\emptyset_f=(2.508-V)/0.499$	$\emptyset$ in degrees
11	A11	Roll Error, Coarse	2.1.1	$\emptyset_c=(2.300-V)/0.0516$	$\emptyset$ in degrees
12	A28	Pitch Error, Fine	2.1.1	$\theta_f=(V-2.501)/1.274$	θ in degrees
13	A12	Pitch Error, Coarse	2.1.1	$\theta_c=(V-2.486)/0.0494$	θ in degrees
14	A14	Scanwheel B Speed	2.1	$R=1712.7 + 73.67V$	R in RPM
15	A43	Full Scale Wheel Speed		$R=338+424V$	R in RPM
16	A15	Roll Bias	2.1.2	$\emptyset_b=0.75V-1.75$	$\emptyset_b$ in degrees

TABLE 1.0-1: CALIBRATION DATA SUMMARY (contd)

NO.	DESIG-NATOR	MEASUREMENT NAME	CALIBRATION DATA REF.	CALIBRATION DATA FORMULA	REMARKS
19	A24	SWA-B H1 Duty Cycle	2.1.1	$\theta_{e1}=35.52V-122.8$	$\phi_{e1}=103.45-29.9V'$
20	A29	SWA-B H2 Duty Cycle	2.1.1	$\theta_{e2}=130.6-37.81V$	$\phi_{e2}=96.84-28.0V$
22	A25	SWA-B Temp.	2.6	$R_T=78700V/(V_{10}-V)$	V_{10} is meas. A41
24	A30	SWA-B Motor Exc.	--	$V_m=6.095V$	V_m in volts
25	A31	Pwr Supply Temp.	2.6	$R_T=78700V/(V_{10}-V)$	V_{10} is meas. A41
26	A26	5 Volt Supply	--	$V_5=V$	V_5 in volts
27	A41	10 Volt Supply	2.1	$V_{10}=6.38+V$	V_{10} in volts
28	A21	Mag. Field Rate, Roll	2.1	$\dot{B}_x=0.00468V-0.0121$	gauss/sec
29	A22	Mag. Field Rate, Pitch	2.1	$\dot{B}_y=0.00476V-0.0123$	gauss/sec
30	A23	Mag. Field Rate, Yaw	2.1	$\dot{B}_z=0.00453V-0.0117$	gauss/sec
31	A20	Spin Accel.	2.1.3	$R=\sqrt{9038V-1808}$	R in RPM
32	E10	+X Solar Pad. Temp. #1	2.6	$R_T=\frac{135,200V}{10.00-V}$	Use Tvs R_T Table
33	E11	+X Solar Pad. Temp #2	2.4, 2.5	$T=39.0V-99.9$	T in °C
34	E8	-X Solar Pad. Temp #1	2.4 2.5	$T=39.0V-99.9$	T in °C
35	E9	-X Solar Pad Temp #2	2.6	$R_T=\frac{135,200V}{10.00-V}$	Use Tvs R_T Table
36	E7	Batt. Temp. #1	2.6	$R_T = 135,200V/(10.00-V)$	"
37	E22	Batt. Temp #2	2.6	$R_T = 135,200V/(10.00-V)$	"
38	E1	Shunt Lim. Current	2.9	$I=1.990V-0.1632$	I in amps

TABLE 1.0-1: CALIBRATION DATA SUMMARY(contd)

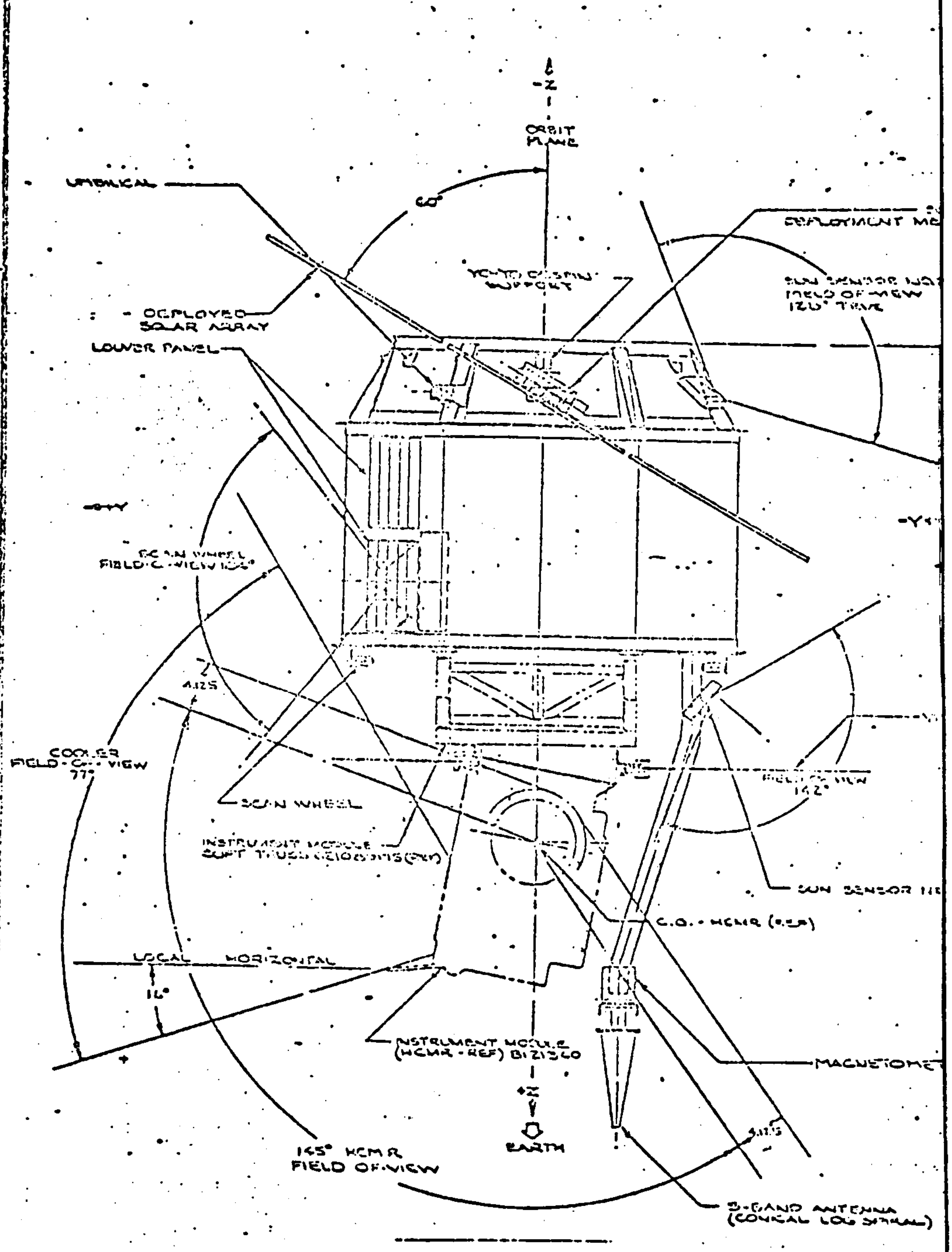
NO.	DESIG-NATOR	MEASUREMENT NAME	REF.	CALIBRATION DATA FORMULA	REMARKS
39	E2	Battery Current	2.9	I=4.02V-10.09	I in amps
40	E3	Main Bus Current	2.9	I=2.002V-0.009	"
41	E4	Reg. Bus Current	2.9	I=0.4909V+0.0191	"
42	E5	Main Bus Voltage	2.9	V_B=6.379V+0.050	V_B in volts
43	E6	Reg. Bus Voltage	2.9	V_r=6.059V-0.200	V_r in volts
44	E15	Solar Array Current	2.9	I_{SA}=1.9976V+0.0113	I in amps
47	T11	S-Band Xmtr Temp	2.6	R_T=135,200V/(10.0-V)	Use Tvs R_T Table (Ref. 6)
48	T20	S-Band Xmtr Power	2.1.2	P=3.596V+0.355	P in Watts
49	T12	VHF Xmtr. Temp	2.6	R_T=135,200V/(10.0-V)	Use Tvs R_T Table
50	T21	VHF Xmtr RF Power	2.14	P=0.06637V+0.0841	P in Watts
55	T14	VHF Rcvr AGC	2.15	S=162V-760 S=4.762V-115	for V $\geq$4.18 for V $\leq$3.10 Interpolate between 3.10 & 4.1
56	T3	A/D Conv. Calib. #1	--	Count No. 5 $\pm$ 1	
57	T4	A/D Conv. Calib. #2	--	Count No. 125 $\pm$ 1	
58	T5	A/D Conv. Calib. #3	--	Count No. 250 $\pm$ 1	

TABLE 1.0-1: CALIBRATION DATA SUMMARY (contd)

NO.	DESIG-NATOR	MEASUREMENT NAME	CALIBRATION DATA REF.	FORMULA	REMARKS
59-85		IM Channels		To be calibrated by GSFC.	
86	S1	Equip. Panel 1 Temp.	2.6	R_T=135,200V/(10.0-V)	Use R_T vs T Table
87	S2	Equip. Panel 2 Temp.	2.6	R_T=135,200V/(10.0-V)	"
88	S3	Equip. Panel 3 Temp.	2.6	R_T=135,200V/(10.0-V)	"
89	S4	Equip. Panel 4 Temp.	2.6	R_T=135,200V/(10.0-V)	"
90	S5	Equip. Panel 5 Temp.	2.6	R_T=135,200V/(10.0-V)	"
91	S6	Scanwheel B Mount Temp.	2.6	R_T=135,200V/(10.0-V)	"
92	S7	Stage 4 Pressure	2.16	P=182.6V+901	Example Only
93	S8	Stage 4 Z Accel	2.16	g=6.2V-1.0	"
94	S9	Stage 4 X Vibration	2.16	g=0.20V-0.50	"
95	S10	Stage 4 Y Vibration	2.16	g=0.20V-0.50	"
96	S11	Stage 4 Z Vibration	2.16	g=0.20V-0.50	"
97	M1	OAS Tank Pressure	2.10	P=99.443V-4.177	P in psia
98	M5	OAS Tank Temp.	2.6	R_T=135,200V/(10.0-V)	Use R_T vs T table
99	M4	OAS Line Temp.	2.6	R_T=135,200V/(10.0-V)	same table
100	M2	OAS Valve Temp.	2.6	R_T=135,200V/(10.0-V)	"
101	M3	Thruster Temp	2.11 2.13	T=0.9925 T_r+242.1V+2.9	T_r is meas. No. 98 (M5)
102-128		Unassigned			

BOEING
Space Systems Division
Spkr: HILL

DELIVERY READINESS REVIEW
HCMM BASE MODULE
2 AUG 1977



ORBITAL CONFIGURATION

Applications
Explorer
Missions
Goddard Space Flight Center

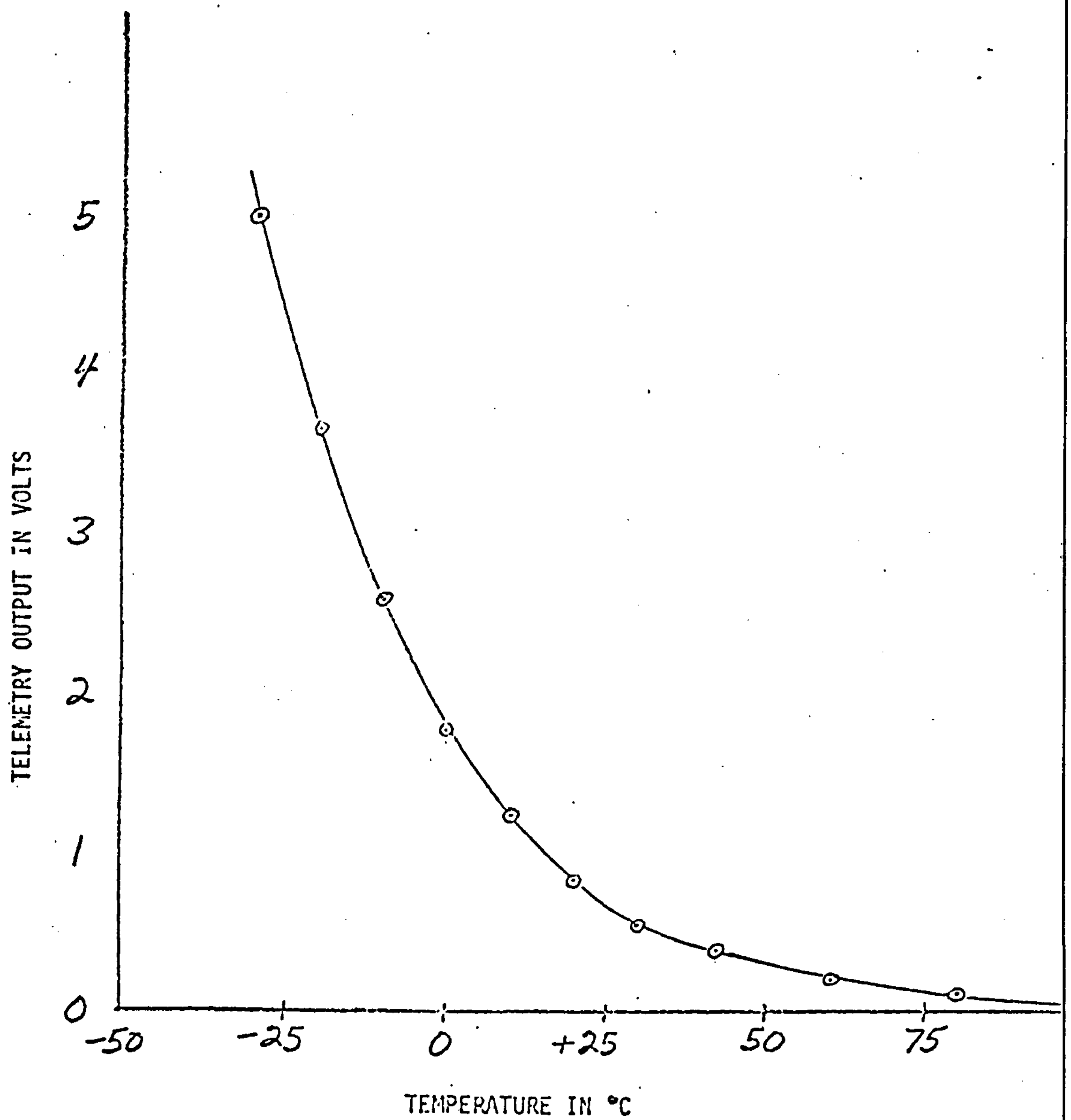


FIGURE 3.4.1: CALIBRATION CURVE FOR YSI 44031 THERMISTOR EXCITED BY PCM ENCODER

REFERENCES

1. G. M. Lerner, "Data Validation and Smoothing," in Spacecraft Attitude Determination and Control, ed. by J. R. Wertz (in preparation)

2. Computer Sciences Corporation, CSC/TR-75/6001, Multisatellite Attitude Determination/Optical Axis Bias Determination (MSAD/OABIAS) System Description and Operating Guide, M. Joseph et al., April 1975

3. ---, CSC/SD-77/6001, SEASAT-A Attitude System Functional Specifications and Requirements, M. Phenneger et al., March 1977

4. Ralph Deutsch, Estimation Theory. Englewood Cliffs, N. J.: Prentice-Hall, Inc., 1965

5. National Aeronautics and Space Administration, Goddard Space Flight Center, NASA X-582-76-77, Mathematical Theory of the Goddard Trajectory Determination System, J. O. Cappellari, C. E. Velez, A. J. Fuchs, April 1976

6. --, GSFC-X-922-74-303, Computation of the IGRF I. Spherical Expansions, B. Trombka and J. Cain, August 1974

7. Computer Sciences Corporation, 3000-17400-01TR, MAPS/GEOS-C Operating Guide and System Description, G. M. Lerner et al., December 1974

8. --, CSC/PR-76/6001, Spacecraft Attitude Determination and Control, ed. by J. R. Wertz (in preparation)

9. Herbert Goldstein, Classical Mechanics. Reading, Massachussetts: Addison-Wesley Publishing Co., Inc., 1959

10. J. D. Lambert, Computational Methods in Ordinary Differential Equations. New York: John Wiley and Sons, Inc., 1973

11. G. M. Lerner, "Magnetometer Models, " in Spacecraft Attitude Determination and Control, ed. by J. R. Wertz, (in preparation)

12. E. Isaacson and H. B. Keller, Analysis of Numerical Methods. New York: John Wiley and Sons, Inc., 1966

13. Computer Sciences Corporation, Applications Explorer Missions-A Heat Capacity Mapping Mission (AEM-A/HCMM) Attitude Determination User's Guide, C. B. Spence et al. (in preparation)